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CONCEPTUAL DESIGN STUDY OF 1985 COMMERCIAL TILT ROTOR TRANSPORTS

Volume III - STOL Design Summary

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Phase I of this study covered V/STOL Tilt Rotor Aircraft. The Design Summary was published in NASA CR-2544, May 1975.

16. Abstract

This report presents results of a conceptual design study of 1985 commercial tilt rotor STOL transports for a NASA 200 n. mi. (370 km) STOL Mission.

The study defines a 100-passenger STOL Variant (Bell D313) of the Phase I VTOL Tilt Rotor Aircraft. Aircraft characteristics are defined; with the aircraft redesigned to meet 2000-Foot (610 m) field criteria, with emphasis on low fuel consumption and low direct operating cost.

The 100-passenger STOL Tilt Rotor Aircraft was analyzed for performance, weights, economics, handling qualities, noise footprint and aeroelastic stability.

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FOREWORD

This report by the Bell Helicopter Company (BHC), Fort Worth, Texas, presents the STOL Design Summary of a conceptual design study of 1985 commercial tilt rotor V/STOL transports. Phase I, in Volumes I and II, presented the results of the VTOL portion of the study. Phase II, in Volumes III and IV, presents the STOL portion. The study is being conducted for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California, under Contract NAS2-8259. Mr. D. R. Brown is the NASA Contracting Officer and Mr. H. K. Edenborough is the NASA Technical Monitor. Mr. K. W. Sambell is the BHC Project Engineer for the study.

The technical guidance of Mr. J. A. DeTore of BHC is especially noted. The assistance and advice of the following members of the BHC technical staff are gratefully acknowledged:

- Mr. B. D. Charles - Aero Acoustics
- Mr. R. D. Foster - Aerodynamics
- Mr. D. A. Hardesty - Handling Qualities
- Mr. E. E. Scroggs, Jr. - Weights
- Dr. J. G. Yen - Aeroelasticity

The BHC tilt rotor aircraft design synthesis methods, available for use on this project, were developed principally by Mr. E. L. Brown. The engine scaling methods were developed by Mr. F. V. Engle.

The volumes prepared are as follows:

- Volume I - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - VTOL Design Summary (BHC Report No. D312-099-002). NASA CR-2544
- Volume II - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - VTOL Substantiating Data (BHC Report No. D312-099-003). NASA CR-137602
- Volume III - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - STOL Design Summary (BHC Report No. D313-099-001).
- Volume IV - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - STOL Substantiating Data (BHC Report No. D313-099-002). NASA CR-137765

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SYMBOLS AND ABBREVIATIONS

AEO	all engines operating
AIA	Aerospace Industries Association
APU	auxiliary power unit
askm	available seat kilometer
assm	available seat statute mile
bh	block hour
BHC	Bell Helicopter Company
BITE	built in test equipment
c.g.	center of gravity
cm	centimeter
C_T	rotor thrust coefficient
cu	cubic
$^{\circ}$, deg	degree
$^{\circ}C$	degrees Celsius
$^{\circ}F$	degrees Fahrenheit
DST	design static thrust
DGW	design gross weight
DOC	direct operating cost
$M	dollars (millions)
FAA	Federal Aviation Authority
FAR	Federal Air Regulation
ft	feet
fpm	feet per minute
fps	feet per second
F/A	fore and aft

SYMBOLS AND ABBREVIATIONS

FS	fuselage station
g	acceleration due to gravity
GW	gross weight
HELO	helicopter
hp	horsepower
hr	hour
IGE	in ground effect
in.	inch
IRP	intermediate rated power (30-min rating)
km	kilometer
kph	kilometers per hour
kt	knot
kw	kilowatt
lb, lbf	pound force
max.	maximum
MCP	maximum continuous power
min	minute
min.	minimum
N	newton
kN	kilonewton
n	normal acceleration
n. mi.	nautical mile
NASA	National Aeronautics and Space Administration
OEI	one engine inoperative
OGE	out of ground effect

SYMBOLS AND ABBREVIATIONS

PAX	passengers
pct, %	percent
PNL	perceived noise level
PNdB	perceived noise level decibels
psf	pounds per square foot
rad	radian
rpm	revolutions per minute
SCAS	stability and control augmentation system
sfc	specific fuel consumption
SL	sea level
SLS	sea level, standard day
ssmpg	seat statute-miles per gallon
std.	standard
s. mi.	statute mile
sq	square
STOL	short takeoff and landing
$T_{1/2}$	time to one-half amplitude
T_2	time to double amplitude
V	velocity
V_1	decision speed for takeoff
V_2	initial climb speed
V_{AP}	approach speed
V_C, V_{MCA}	minimum control airspeed
V_{CON}	Airspeed at which transition is complete and the aircraft enters the aerodynamic flight regime.

SYMBOLS AND ABBREVIATIONS

V_D	dive speed
V_{LOF}	minimum flying speed
V_{MCG}	minimum control ground speed
V_t	total airspeed
θ_m	mast angle (airplane: zero^{o})
σ	rotor solidity ratio
σ'	atmospheric density ratio
ψ	yaw acceleration
ω_n	undamped natural frequency

1. SUMMARY

This report presents the results of a conceptual design study of a 1985 commercial STOL tilt rotor transport based on a NASA 200 n. mi. (370 km) STOL mission. The purpose of the study is to generate transport designs to support V/STOL transportation system studies by NASA.

Phase I of the study, which was published in May 1975, defined a 45-passenger VTOL tilt rotor aircraft (Bell D312) based on the generic characteristics of the NASA-ARMY XV-15 Tilt Rotor Research Aircraft.

Phase II of the study, reported in this volume, defines an STOL variant of the Phase I VTOL tilt rotor. Aircraft characteristics are defined, with the aircraft redesigned to meet 2000-foot (610 m) field criteria, with emphasis on low fuel consumption and low direct operating cost.

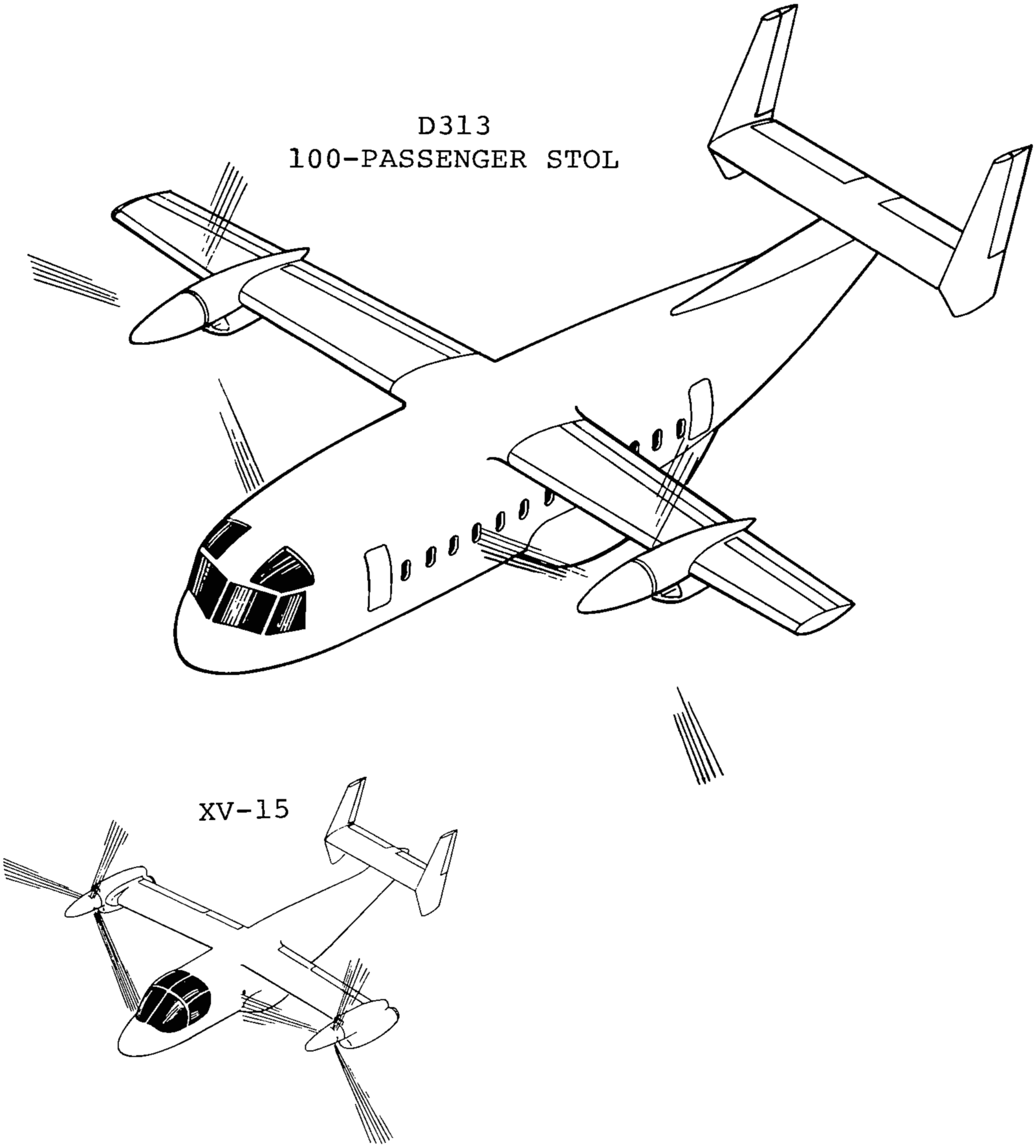
The selected STOL design approach was to retain the same installed power and design range as the 45-passenger VTOL aircraft. The engines, rotors and transmissions are identical. Design changes included the addition of a high aspect-ratio, high-lift wing. The payload capability increased to 100 passengers (study maximum). The resulting aircraft (Bell D313) is shown in Figure 1-1, to the same scale as the XV-15. Compared to the 45-passenger VTOL, the direct operating cost decreased 43% and the fuel economy improved by 137% (from 34.2- to 81.1 seat-miles per gallon).

The 100-passenger STOL tilt rotor aircraft was analyzed for performance, weights, economics, handling qualities, noise footprint and aeroelastic stability. Significant results are shown in Table 1-1.

In addition, at fuel costs of 10¢/lb (probably conservative for 1985) the 1985 STOL tilt rotor is considered to have comparable direct operating costs to the 1985 CTOL turbo-fan at 200 n. mi. (370 km) range. (See Appendix)

By using the same type of gimbal-mounted tilt rotor system that will be tested on the XV-15, the STOL version of the tilt rotor aircraft is predicted to achieve the unique capability of making zero-crab, zero-bank STOL approaches in 25-knot crosswinds. This capability is a potential item for verification during future flight simulator experiments and advanced research flight testing with the XV-15.

FIGURE 1-1
COMPARISON OF XV-15 AND D313



IDENTICAL SCALES

TABLE 1-1

100-PASSENGER STOL TILT ROTOR AIRCRAFT CHARACTERISTICS, BELL D313

NASA MISSION: 200 N.M. DESIGN RANGE

DESIGN FIELD: 2000 FT, SEA LEVEL 90°F
35 FEET CLEARANCE HEIGHT

ITEM	UNIT	VALUE
Noise at 500 ft. sideline, takeoff	PNdB	96.0
Direct Operating Cost, @ 200 n.m.	¢/assm (¢/askm)	2.67 (1.66)
(per available seat @ 400 n.m. statute mile)	¢/assm (¢/askm)	2.37 (1.47)
Area of 95 PNdB Contour, Takeoff	acres (km^2)	113.4 (.459)
Area of 95 PNdB Contour, Landing	acres (km^2)	57.9 (.234)
Rotor Diameter	ft (m)	43.6 (13.29)
Design Gross Weight	lbf (kN)	64300 (286.02)
Weight Empty	lbf (kN)	42720 (190.03)
Installed Horsepower, (Total, 30-Min. Rating, SLS)	hp (kw)	9072 (6765)
Disc Loading (Based on Thrust)	psf (N/m^2)	16.15 (773.1)
Wing Loading	psf (N/m^2)	72.5 (3471)
Hover Tip Speed	fps (m/s)	700 (213.4)
Cruise Tip Speed	fps (m/s)	600 (182.9)
Block Fuel	lbf (kN)	1888 (8.398)
Block Time, Engines-On	hrs	1.015
Cruise Speed at 20000 Feet, (6096 m) Std. Day	knots (kph)	248 (459)

2. INTRODUCTION

Low disc loading aircraft have high efficiency to convert energy to thrust, with ample control, at low speeds. The low disc loading tilt rotor aircraft combines these low speed attributes with efficient and quiet flight in airplane mode.

The VTOL tilt rotor aircraft concept will be investigated by NASA and the Army, with the XV-15 Tilt Rotor Research Aircraft. The XV-15 is currently being fabricated by Bell Helicopter Company (BHC), Reference 2-1, with first flight scheduled for 1976. Characteristics of VTOL Tilt Rotor aircraft for commercial service in 1985 have been studied by Bell Helicopter Company during the first phase of this study, References 2-2, 2-3, and by Boeing, Reference 2-4. The BHC study resulted in the definition of a 45-passenger tilt rotor transport, designated the D312.

Since tilt rotor aircraft have the capability to lift increased payloads when runways are available, the study was extended to this second phase to define the characteristics of tilt rotor aircraft when designed specifically for STOL operations. Boeing included results of their STOL tilt rotor study in Reference 2-4.

The improved lifting efficiency of tilt rotor aircraft in STOL operations can be utilized in three ways:

a. Increased payload and design gross weight at the same design range and engine size.

b. Increased design range at the same payload, design gross weight and engine size.

c. Reduced engine size and design gross weight at the same payload and design range.

Bell analyzed these three approaches and assessed each one from the viewpoints of fuel economy and direct operating cost. Approach a. was selected for design verification. This included analysis of performance, weights, economics, handling qualities, noise footprint and aeroelastic stability.

The NASA Study Guidelines and Constraints, Reference 2-5, are summarized, for the STOL phase, in Table 2-1.

TABLE 2-1
STUDY CONSTRAINTS AND GUIDELINES

NASA 1985 COMMERCIAL STOL TILT ROTOR TRANSPORT STUDY

NASA CONTRACT STATEMENT OF WORK:

"...DEFINE AIRCRAFT CHARACTERISTICS IF THE BASELINE TILT ROTOR AIRCRAFT OF PHASE I IS REDESIGNED AS AN STOL AIRCRAFT TO MEET 2000-FOOT FIELD CRITERIA.
EMPHASIS ON:
- ECONOMICS
- LOW FUEL CONSUMPTION"

CONSTRAINTS: MAXIMUM PAYLOAD OF 100 PASSENGERS

DESIGN GUIDELINES:

- MISSION
  DESIGN FIELD LENGTH SL 90°F, CLEARANCE HEIGHT 35 FEET
  200 N. MI. RANGE + 50 N. MI. ALTERNATE LEG + LOITER

- PAYLOAD
  180 LB/PASSENGER, INC. BAGGAGE
  190 LB/CREWMAN, INC. GEAR
  140 LB/CABIN ATTENDANT, INC. GEAR

- FUSELAGE
  DOUBLE AISLE

- EQUIPMENT
  2100 LB + SEATS

- TECHNOLOGY LEVEL
  25% WEIGHT REDUCTION FROM PRESENT
  - BODY, EMPENNAGE, WING
  - ENGINE NACELLES
  - FLIGHT CONTROLS (NONROTATING)

- ENGINES
  NASA-DEFINED CRITERIA
  FUEL SFC = 0.42 LB/SHP HR,
  TOP @ SL 90°F
  SPECIFIC WEIGHT = 0.15 LB PER SHP

- STABILITY & CONTROL
  NASA-DEFINED CRITERIA

- ECONOMICS
  NASA-DEFINED UNIT COSTS FOR INITIAL COST
  NASA-DEFINED AIA METHOD FOR DOC

3. APPROACH

3.1 GENERAL

The NASA study guidelines required the design point VTOL tilt rotor, which was selected in Phase I of this study (Bell D312), to be redesigned specifically for STOL operations for a 2000-foot (610 m) field length. Hover performance was not required and emphasis was placed on achieving low fuel consumption and low direct operating cost. After initial study, the following design changes were made to optimize for STOL:

- The wing aspect-ratio was increased from 6.86 to 10.0 to improve specific range.

- The wing maximum lift coefficient was increased from 2.0 to 3.0 by the use of a full span Fowler trailing edge flap and a leading edge slat. These changes increased lifting efficiency during the rolling takeoff.

- Full span spoilers were incorporated to dump excessive wing-lift during high descent rates on the approach. The outer segment is used for roll control in airplane mode.

- The wing loading was reduced from 80 psf (3.83 kN/m^2) to 72.5 psf (3.47 kN/m^2) to match the slightly slower cruise speed of the STOL aircraft.

- The pylon conversion axis was moved from 55% mac to 5% mac (ahead of the wing forward spar) to reduce tail download during fuselage rotation at takeoff. This also enabled the wing span to be extended outboard of the rotor center-line. Thus the rotor blade/fuselage clearance was retained and the wing root bending loads (produced by rotor thrust) were held at the original levels of the D312.

- The wing sweep angle was reduced from -6.5 degrees to zero. This simplified the wing design for the wing span-extension outboard of the rotor axes.

- The landing gear was moved forward to minimize tail down-load during rotation at takeoff.

- The fuselage nose-up ground clearance angle was increased from 10.0 degrees to 15.0 degrees, to allow for steeper flares required by STOL operations.

With these design changes, preliminary mission iteration proceeded to establish aircraft solutions. These solutions had common (or generic) characteristics which are shown in Table

3.1-1. The critical mission parameter was established to be the landing distance. Takeoff distance was not critical because of the interconnected rotors and the emergency (2½-minute) engine power available (1.2 x 30-minute rating) with one engine inoperative.

With these generic characteristics defined, aircraft synthesis proceeded to establish solutions for each of the three STOL design variants of the 45-passenger VTOL D312 aircraft:

- I constant payload and range
- II constant installed power and range
- III constant installed power and payload

The aircraft characteristics for the solutions of each approach are shown in Table 3.1-2. Approach II, which resulted in a 100-passenger design, was selected because of its high fuel economy of 81.1 seat-miles per gallon (29.60 seat-kilometers per liter) and low direct operating cost of 2.67 ¢/assm (1.66 ¢/askm). This design was designated the D313. In addition to the engines, the rotors and transmissions are identical to those of the D312 VTOL aircraft. Thus, there is considerable dynamic system commonality possible between a 100-passenger STOL tilt rotor aircraft and a 45-passenger VTOL tilt rotor aircraft. An isometric comparison at the same scale is shown in Figure 3.1-1.

3.2 OPTIMUM DISC LOADING INVESTIGATION

With the STOL aircraft "design-approach" selected as shown by approach II in Table 3.1-2, the question of the optimum disc loading remained.

Aircraft solutions (100-passenger class, 200 n. mi., 370 km range) were synthesized with design disc loadings of 16.15, 18.30 and 20.46 psf. The design thrust/gross weight ratio was held at 0.75 (to determine power loading) and the remaining generic characteristics of Table 3.1-1 were retained. A summary of results is shown in Table 3.2-1. It was found that as disc loading increased, the lifting efficiency (DGW/Installed Power) reduced and the fuel economy (seat-miles per gallon) also reduced. The final DOC analysis should be done with one specific engine size specified for all aircraft and this was beyond the scope of this study. However, if say, 10000 shp was specified, the following are the design point solutions:

Design Disc Loading psf	Design Disc Loading kN/m^2	Design Gross Weight lbf	Design Gross Weight kN
16.15	(.773)	70,900	(315.4)
18.30	(.876)	67,300	(299.4)
20.46	(.979)	63,500	(282.5)

TABLE 3.1-1
GENERIC CHARACTERISTICS OF STOL COMMERCIAL TILT ROTOR AIRCRAFT,
2000 FT FIELD AT S.L. 90°F

T/W (AT C_T/SIGMA = 0.124)	0.75
DISC LOADING (BASED ON THRUST)	16 psf
WING LOADING	70-85 psf
WING ASPECT RATIO	10
TRANSMISSION SIZING CRITERIA	Develop Design Static Thrust at S.L. 90°F
ENGINE SIZING CRITERIA (4 ENGINE AIRCRAFT)	Develop Design Static Thrust With One Engine Out. Remaining Engines at Emergency Rating
POWER LOADING	7 Pounds Weight/hp
CRUISE SPEED	225-300 kt
CRUISE ALTITUDE	20,000 ft
CROSS WIND CAPABILITY	25 kt on 80 kt Approach Zero Bank and Crab

TABLE 3.1-2
STOL TILT ROTOR AIRCRAFT CHARACTERISTICS

PARAMETERS HELD CONSTANT →	VTOL BASELINE	STOL I PAYLOAD & RANGE	STOL II POWER & RANGE	STOL III POWER & PAYLOAD
DGW, lbf	44848	36975	64300	64300
PAYLOAD, No. Passengers	45	45	100	45
RANGE, n.mi.	200	200	200	2402
ENGINE RATING[1], hp (30-MIN, S.L.S.)	2268	1311	2268	2268
SPEED, knot	297[(2)]	234[(3)]	248[(3)]	227[(3)]
CRUISE ALTITUDE, feet	11000	20000	20000	20000
FUEL ECONOMY, ssmpg	34.2	61.0	81.1	51.3
DOC, ¢/assm (@ Fuel Cost $.02/lb)	4.66	4.54	2.67	4.31

[1]FOUR ENGINES

[2]AVERAGE CRUISE SPEED AT 90% MAXIMUM CONTINUOUS POWER

[3]AVERAGE CRUISE SPEED AT 99% MAX RANGE

FIGURE 3.1-1
COMPARISON OF D312 AND D313 TILT ROTOR AIRCRAFT

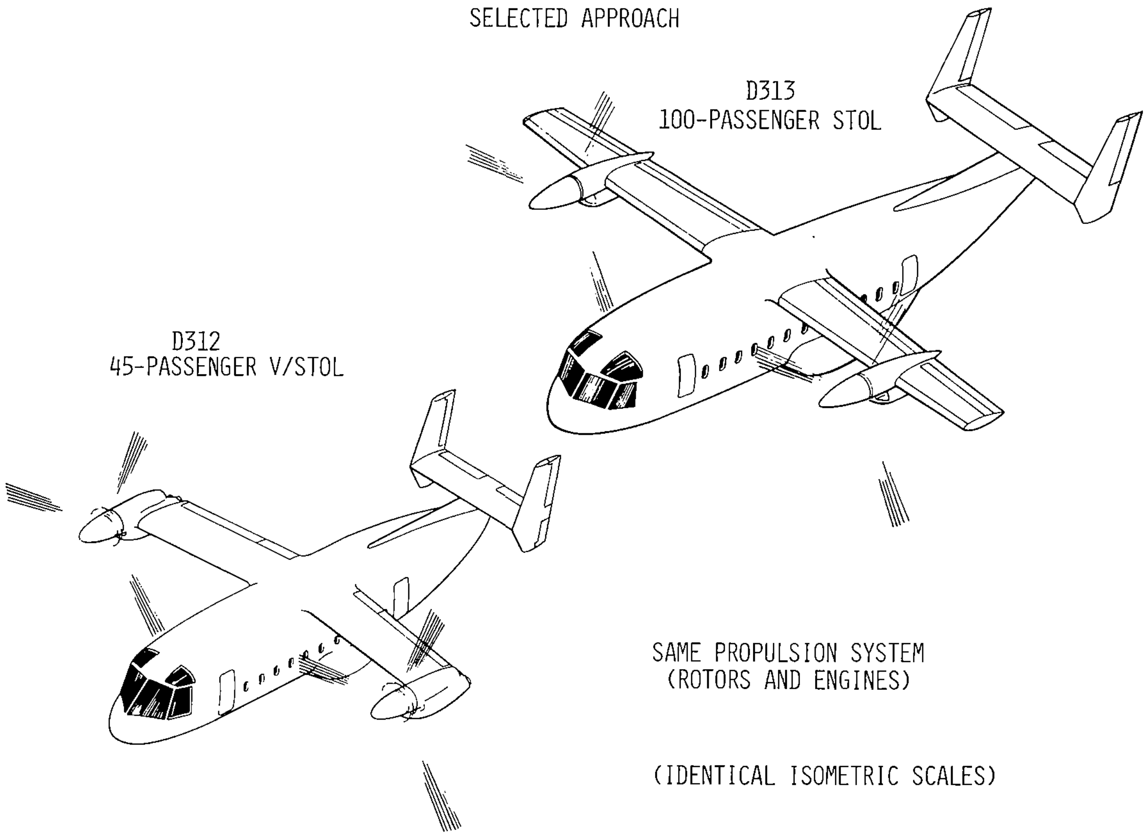


TABLE 3.2-1
DISC LOADING, LIFTING EFFICIENCY AND FUEL ECONOMY

DISC LOADING,	psf	16.15	18.30	20.46
DGW/INSTALLED POWER,	lbf/hp	7.09	6.73	6.35
FUEL ECONOMY,	ssmpg	81.1	81.0	80.6

CONCLUSIONS

- DISC LOADING OF 16.15 PSF SELECTED TO MAXIMIZE LIFTING EFFICIENCY AND FUEL ECONOMY
- FINAL DOC ANALYSIS FOR OPTIMUM DISC LOADING NEEDS A SPECIFIC ENGINE SIZE TO BE SELECTED --- BEYOND THE SCOPE OF THIS STUDY

The higher disc loading solution would cruise slightly faster but the increased payload carried by the lower disc loading solution would produce a lower direct operating cost.

Thus the lower disc loading solution is recommended because of its higher lifting efficiency, higher fuel economy and lower direct operating cost.

The 100-passenger class, 200 n. mi. (370 km) range, D313 is described in the next section.

4. DESCRIPTION OF SELECTED AIRCRAFT

4.1 GENERAL

A three view of the selected D313 STOL aircraft is shown in Figure 4.1-1. The significant characteristics are the low disc loading rotors and the high aspect-ratio wing.

The generic characteristics of the XV-15 tilt rotor system have been retained. The three bladed stiff-in-plane tilt rotors have a design disc loading (based on thrust) of 16.15 psf (.773 kN/m^2). Takeoff and cruise tipspeeds are 700 ft/sec (213 m/sec) and 600 ft/sec (183 m/sec) respectively. Gimbal hubs provide relief for one-per-rev flapping airloads (and virtually eliminate Coriolis forces induced by flapping) which reduces inplane bending moments. A moderate amount of hub restraint is used to increase control power and damping in helicopter mode without generating high blade loads. Flapping clearance is 12 degrees and fuselage/blade-tip clearance is 12 inches (.3 m).

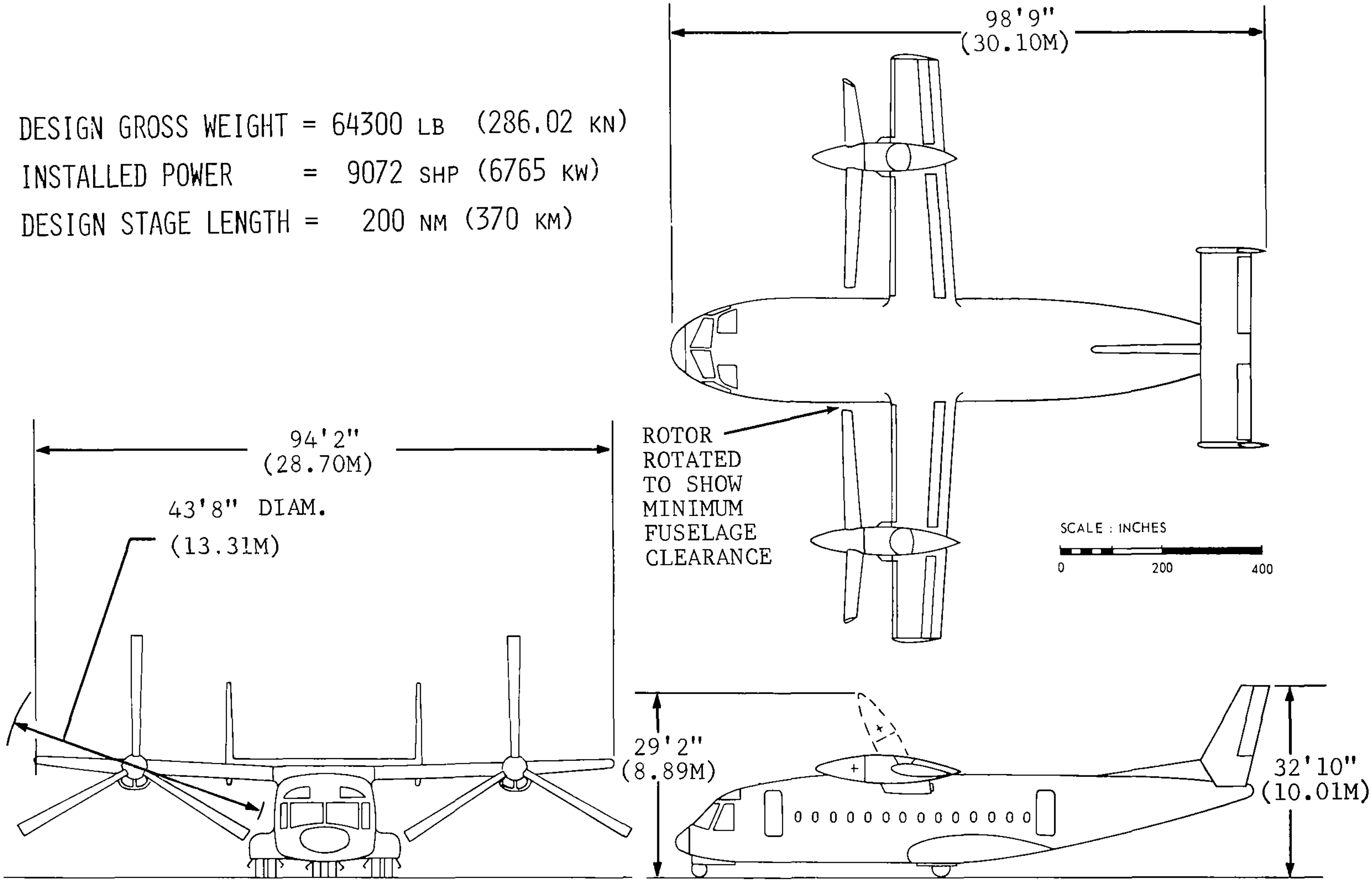
The zero-sweep wing has an aspect ratio of 10.0 and a taper ratio of 1.37. Wing loading is 72.5 psf (3.47 kN/m^2) and maximum wing lift coefficient is 3.0. The wing has a constant 23% thickness chord ratio and is fitted with the following lift/control devices:

- Trailing-edge Fowler flap of 29% chord, full span.
- Leading-edge slat of 10% chord, full span.
- Spoilers of 15% chord, pivoted at 60% mac, full span. (The spoiler segment outboard of the pylons is used for roll control in airplane mode.)

The pylon conversion axis is at 5% mac. The aft end of the pylon fairing is nonconverting and a spring-loaded cover provides a smooth fairing between the converting and non-converting portions. Generally, the rotor and wing controls are installed aft of the wing box and the interconnect shaft is installed forward of the wing box.

The four turboshaft engines are mounted in pairs on the rotor pylons. High transmission efficiency is possible since the normal rotor drive is via herringbone and planetary gears. The rotors are mechanically interconnected so that any engine can power either rotor.

FIGURE 4.1-1
STOL TILT ROTOR TRANSPORT, 100-PASSENGER



Fuselage pressurization is provided to hold cabin pressure at the equivalent of 3000 ft (914 m) pressure altitude. In normal operation, this results in a cabin pressure rate-of-change not exceeding the equivalent of a descent rate of 300 fpm (91.4 m/min).

The H-configuration empennage of the XV-15 is retained. The 100-passenger fuselage has different fuselage pitch and yaw moment coefficients than those of the XV-15. Based on stability analyses allowing for these effects, the D313 horizontal tail volume coefficient is 1.639 and the vertical tail volume coefficient is 0.130.

The body is sized by the NASA Study Guidelines and Design Criteria and provides airline passenger accommodations with a double aisle. Passenger checked baggage volume, 2.5 cu ft (0.071 cu m) per passenger, is provided in the fuselage belly. These guidelines led to the noncircular fuselage cross sections shown. Additional overall system studies should investigate fuselage belly requirements to carry mail/freight and, if so, a circular cross section could be justified.

The cockpit has excellent visibility for V/STOL operations. Downward visibility of 25 degrees is provided for steep approaches, if necessary. Typical fuselage attitude on an STOL final approach is $+3.5^{\circ}$ on a 6° glideslope.

The landing gear is designed for rolling takeoff and landing at speeds up to 80 knots (148 kph). Tip-over angle is a minimum of 27° laterally and 20° longitudinally.

4.2 FUSELAGE LAYOUT

The 100-passenger fuselage layout, shown in Figure 4.2-1, has seven-abreast seating. As required by the NASA guidelines, the following are provided: two doors, two aisles, space for two cabin attendants, two lavatories, beverage service, coat rack, ticket center, and built-in air stair. In the fuselage belly, baggage compartments are provided to allow 2.5 cu ft (.071 cu m) per passenger. These accommodation requirements were adequately met by a noncircular cross section. The fuselage external width and height is 200 inches (5.08 m) and 170 inches (4.32 m) respectively, and the overall length is 1150 inches (29.21 m).

4.3 DESIGN POINT MISSION ANALYSIS

The NASA mission was represented by 21 segments which allowed for the basic 200 n. mi. leg, the 50 n. mi. alternate leg, and the 20-minute hold. A mission schematic is shown in Figure 4.3-1. Engine fuel flow estimation was based on matching the NASA reference point at sea level $90^{\circ}F$ ($32.2^{\circ}C$) with typical

FIGURE 4.2-1
FUSELAGE LAYOUT, 100-PASSENGER

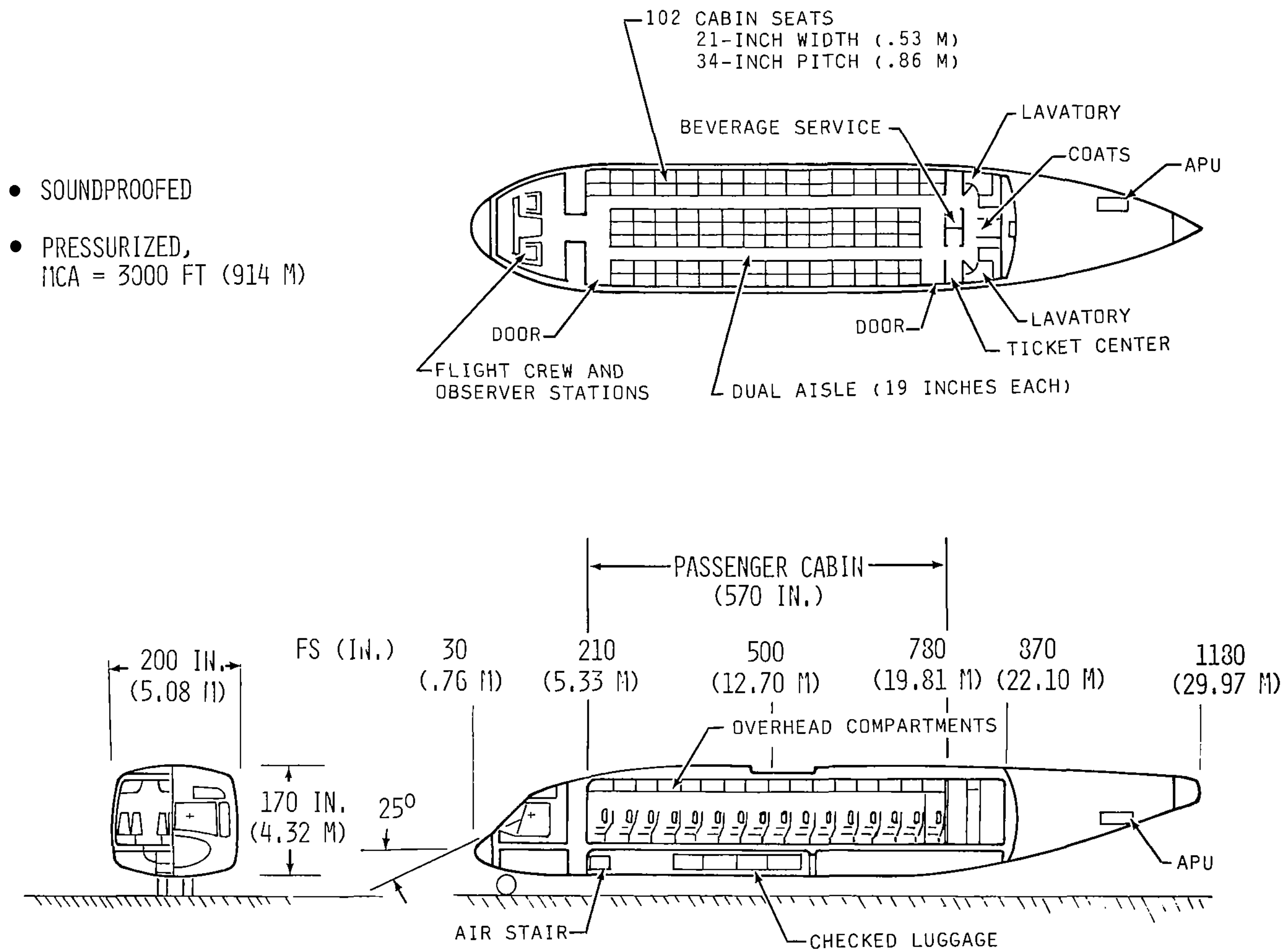
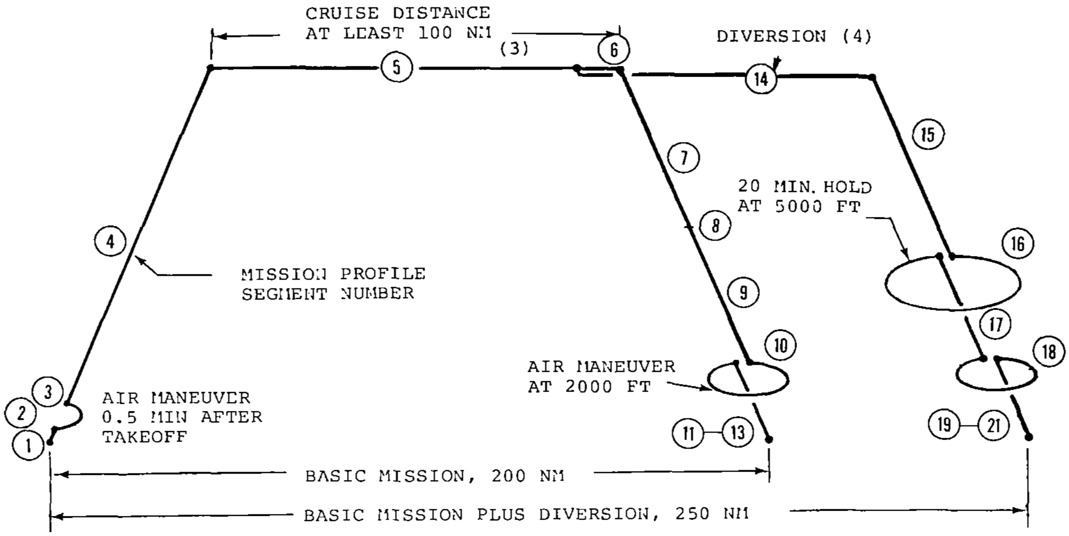


FIGURE 4.3-1
NASA 200 NM STOL MISSION PROFILE



NOTE
1. CRITICAL SIZING FOR TAKEOFFS AND LANDING AT SL 90°F.
2. MISSION FUEL ANALYSIS FOR STANDARD DAY.
3. CRUISE ALTITUDE AND SPEED SELECTED USING MAXIMUM SPECIFIC RANGE AS A GUIDE.
4. DIVERSION AT SPEED FOR BEST RANGE AT CRUISE ALTITUDE.

engine technology (Reference 4-1) of the 1980-85 time frame. The BHC-defined items of the mission profile: climb speed (1.2 V stall), cruise speed (99% maximum range) and cruise altitude (20000 feet) were determined from minimum mission fuel requirements. Other combinations were explored such as: climb speed (1.8 V stall), cruise speed (90% maximum continuous power) and cruise altitude (11000 feet) but the fuel used increased by up to 6%.

4.3.1 MISSION SEGMENT ANALYSIS - Results of the 200 n. mi. (370 km) mission analysis for the D313 are shown in Table 4.3-1. Calculations are shown for time required, distance covered, and fuel required for 21 mission segments. Cumulative values are shown for the mission status at the end of each segment. Significant results are that the 100-passenger D313 used only 1888 lbf (8.398 kN) of fuel. This is 81.1 seat-miles per gallon (29.60 seat-kilometers per liter) and illustrates the fuel efficiency of advanced technology STOL tilt rotor aircraft. (For reference, present turbofan aircraft use 2.5-3.0 times this amount of fuel for the same mission.) Reserve fuel was 877 lbf (3.901 kN) or 46% of fuel consumed.

4.4 GROUP WEIGHT STATEMENTS

The NASA guidelines allowed a 25% weight reduction from present technology for the following components: body, empennage, wing, engine nacelles and nonrotating flight controls. The BHC weight estimating method was based on the following:

Rotor Group - Actual weights for the XV-15 rotor group, detailed design study of the Bell Model 266 tilt rotor (DGW = 28,000 lbf) and general helicopter experience.

Drive System - General helicopter experience at BHC.

Wing Group - Analytical method based on calculated design conditions. No statistics were found to be applicable to wings for tilt rotor aircraft.

Engine Group - Basic engine specific weight was defined by the NASA Study Guidelines and is considered to be representative of 1980 technology.

Body Group - Commercial airliner statistical data.

All other components and systems were based on statistical weight data available to BHC.

The group weight statement for the D313 is shown in Table 4.4-1. The empty weight is 42720 lbf (190.0 kN) and the empty weight ratio is 0.664.

TABLE 4.3-1
MISSION SEGMENT ANALYSIS, 100-PASSENGER STOL AIRCRAFT

SEGMENT	MODE	WP	H	V	D	T	W	FR
	(START)	18660	0	0	0	0.000	64300	0
1	WUP	18660	0	0	0	0.017	64265	35
2	TOF	18660	0	0	0	0.033	64214	86
3	ALO	18660	0	141	0	0.050	64189	111
4	ACL	18660	20000	180	33	0.256	63626	674
5	ACR	18660	20000	248	170	0.807	62596	1704
6	ACR	18660	20000	248	171	0.811	62589	1711
7	DSC	18660	10000	248	187	0.878	62551	1749
8	ACR	18660	10000	222	188	0.882	62543	1757
9	DSC	18660	2000	222	200	0.936	62513	1787
10	ALO	18660	2000	139	200	0.961	62476	1824
11	DSC	18660	1000	139	200	0.977	62464	1836
12	DSC	18660	0	139	200	0.998	62446	1854
13	GND	18660	0	0	200	1.015	62412	1888
14	ACR	18660	20000	248	227	1.038	62167	2133
15	DSC	18660	5000	248	248	1.122	62121	2179
16	ALO	18660	5000	145	248	1.455	61643	2657
17	DSC	18660	2000	145	250	1.471	61636	2664
18	ALO	18660	2000	138	250	1.496	61599	2701
19	DSC	18660	1000	138	250	1.513	61588	2712
20	DSC	18660	0	138	250	1.534	61570	2730
21	WUP	18660	0	0	250	1.551	61535	2765

(12.30 KN)

MODES:
- ACL - AIRPLANE CLIMB
- ACR - AIRPLANE CRUISE
- ALO - AIRPLANE LOITER
- DSC - DESCENT
- GND - GROUND OPERATION
- TOF - STOL TAKEOFF
- WUP - WARMUP

- MISSION FLOWN ON STANDARD DAY
- FOR RESERVE LEG: INITIAL CONDITIONS OF SEGMENT 14 EQUAL FINAL CONDITIONS SEGMENT 5.

TABLE 4.4-1
GROUP WEIGHT STATEMENT, 100-PASSENGER STOL AIRCRAFT

ROTOR GROUP			4296 LBS
WING GROUP			5149
TAIL GROUP			691
HORIZONTAL		396	
VERTICAL		295	
BODY GROUP			7871
LANDING GEAR			2730
NOSE		646	
MAIN		2045	
AUXILIARY		39	
FLIGHT CONTROLS GROUP			3530
NONROTATING		2597	
ROTATING		481	
CONVERSION SYSTEM		452	
ENGINE SECTION			633
PROPULSION GROUP			7835
ENGINE INSTALLATION		1712	
EXHAUST SYSTEM		96	
LUBRICATION SYSTEM		336	
FUEL SYSTEM		190	
ENGINE CONTROLS		244	
STARTING SYSTEM		134	
DRIVE SYSTEM		5123	
GEARBOXES	4396		
SHAFTING	727		
INSTRUMENT GROUP			293
HYDRAULIC GROUP			416
ELECTRICAL GROUP			495
AVIONICS GROUP			458
FURNISHINGS AND EQUIPMENT GROUP			5917
ENVIRONMENTAL CONTROL GROUP			2066
AUXILIARY POWER UNIT			338
OTHER			0
LOAD HANDLING GROUP			0
WEIGHT EMPTY			42720 LBS
			(190.03 KN)

TABLE 4.5-1
MISSION WEIGHT SUMMARY, 100-PASSENGER STOL AIRCRAFT

WEIGHT EMPTY	42720 LBS
CREW	660
PAYLOAD	18000
AUXILIARY TANK	0
TRAPPED FLUIDS	154
FUEL AVAILABLE	2766
MISSION GROSS WEIGHT	64300
DESIGN GROSS WEIGHT	64300 LBS
	(286.02 KN)

4.5 MISSION WEIGHT SUMMARY

The mission weight summary for the D313 100-passenger aircraft is shown in Table 4.5-1. Crew and passenger weights are per NASA guidelines:

Pilot Crew (2)..........190 lbf (845 N), each, including gear
Cabin Attendant (2).....140 lbf (623 N), each, including gear
Passengers (100) ....180 lbf (801 N), each, including baggage

4.6 ECONOMICS

The economic analysis was based on NASA guidelines and the 1968 Aerospace Industries Association method to estimate direct operating costs, Reference 4-1. This approach to economics is considered by BHC to be adequate at the conceptual design stage. The AIA method estimates the DOC of V/STOL aircraft by allowing for the initial cost and weight of the dynamic systems and then adding this to the airframe and engine costs. BHC compared the AIA method to BHC methods used in Reference 4-2 and found good correlation. It should be noted that if the AIA method was used on an alternative V/STOL concept with a large number of small components, but which had the same total weight and initial cost as the tilt rotor aircraft, then the maintenance cost predicted would be the same and, therefore, would probably be optimistic for the alternative concept.

The D313 was analyzed for initial cost and direct operating cost for the NASA design mission with climb rates, cruise speeds and altitude selected to minimize fuel consumption as described in Section 4.3 "DESIGN POINT MISSION ANALYSIS".

The following cost data were used:

- airframe cost, $90* and $110 per pound
- dynamic system cost, $80 per pound
- utilization, 2500* and 3500 block hours per year
- depreciation period, 12 years

*"baseline-cost" conditions

The avionics group cost ($0.25M) has been included in the initial cost and in the depreciation cost, but it has not been included in the airframe maintenance cost equations. All other costs were computed per NASA guidelines and the AIA cost method.

4.6.1 FIRST COST AND DIRECT OPERATING COST - Table 4.6-1 shows first cost and direct operating cost at the design range of 200 n. mi. (370 km) for variations in assumed utilizations and airframe unit costs.

4.6.2 DIRECT OPERATING COST VERSUS RANGE - Figure 4.6-1 shows direct operating cost from 100 n. mi. (185 km) to 500 n. mi. (926 km) for the baseline cost conditions. At ranges up to 200 n. mi. (370 km) the aircraft cruised at 99% of best range speed (248 knots, 459 kph). For ranges above 200 n. mi. (370 km), extra fuel capacity was installed and payload was reduced to keep takeoff weight at design gross weight; also, at these higher ranges it was found that significantly better DOC was achieved by cruising at 90% of maximum continuous power, 255 knots.

The minimum DOC was 2.37 ¢/assm (1.47 ¢/askm) at 435 n. mi. (806 km) range and with a payload of 90 passengers.

Table 4.6-2 shows direct operating cost from 50 statute miles (80 km) to 500 statute miles (805 km) for the baseline cost conditions.

4.6.3 FUEL ECONOMY VERSUS RANGE - Figure 4.6-2 shows the fuel economy index as measured in seat statute-miles per gallon (ssmpg) for ranges up to 500 n. mi. (926 km). The D313 achieves 81.1 ssmpg at 200 n. mi. (370 km) and a maximum of 82.0 ssmpg at 330 n. mi. (611 km) indicating the suitability of the tilt rotor aircraft for a fuel conservative design approach.

4.6.4 FUEL ECONOMY VERSUS DESIGN CRUISE SPEED - Figure A-2 (Appendix) shows the mission fuel economy index versus design cruise speed at a range of 200 n.mi.. At each cruise speed the fuel economy index is shown for the mission solution, with takeoff at Design Gross Weight. Maximum fuel economy is 81.7 ssmpg at a design cruise speed of 227 knots (420 kph). At higher speeds the fuel economy drops to 72.0 ssmpg at a design cruise speed of 300 knots (556 kph).

TABLE 4.6-1
DIRECT OPERATING COST VERSUS UTILIZATION AND AIRFRAME COST, 100-PASSENGER STOL AIRCRAFT

UTILIZATION, BH/YR	AIRFRAME COST $/LB	FIRST COST INC. SPARES, $M	DOC @ 200 N.MI., ¢/ASSM (¢/ASKM)
2500	90	4.899	2.67 (1.66)
2500	110	5.524	2.80 (1.74)
3500	90	4.899	2.42 (1.50)
3500	110	5.524	2.52 (1.57)

FIGURE 4.6-1
DIRECT OPERATING COST VERSUS RANGE, 100-PASSENGER STOL AIRCRAFT

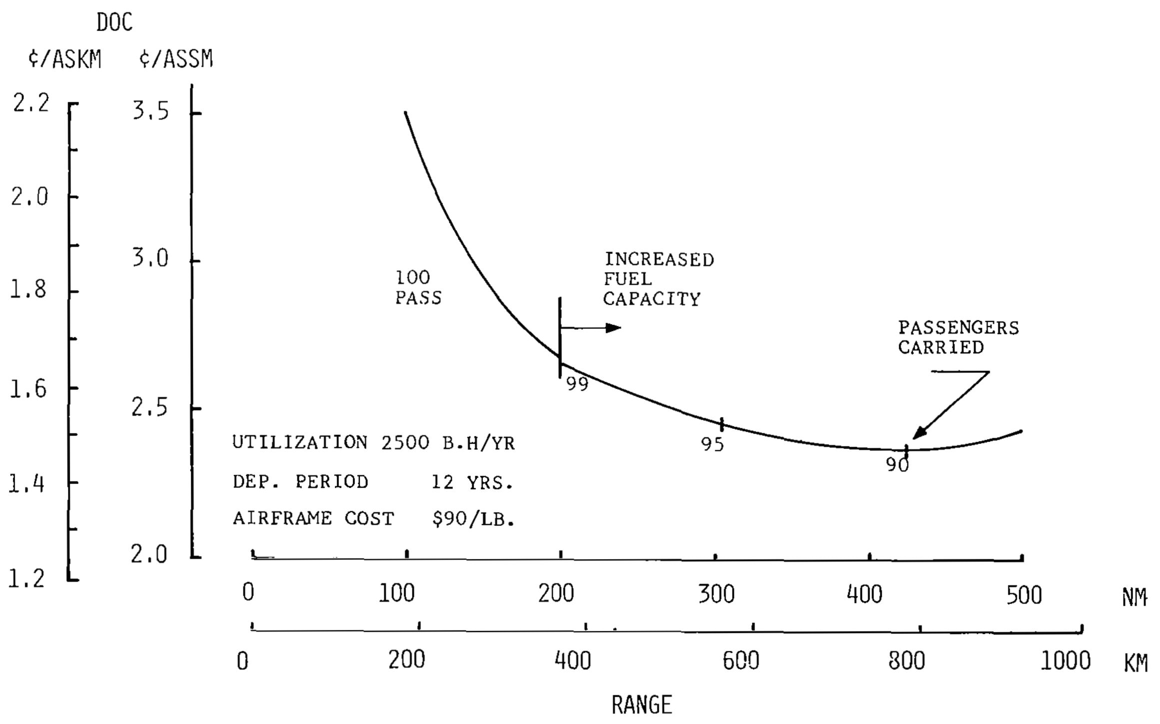


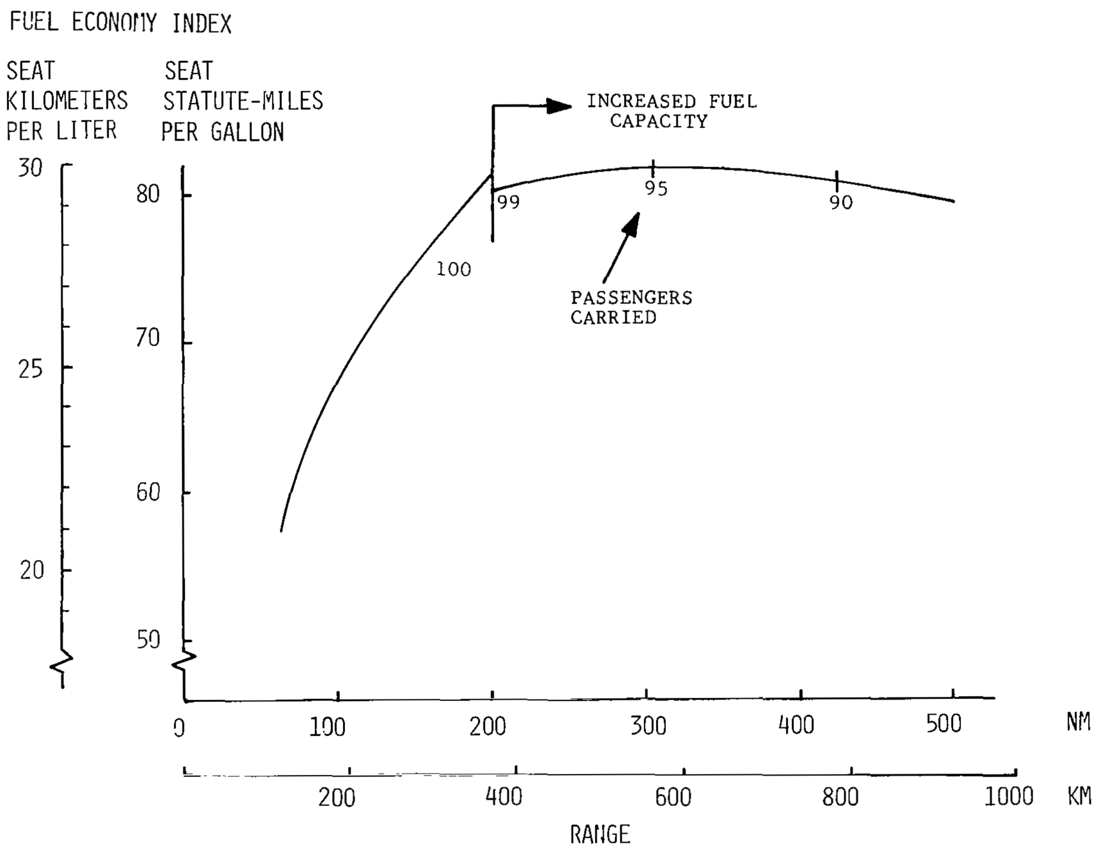
TABLE 4.6-2
DIRECT OPERATING COST VERSUS RANGE, 100-PASSENGER STOL AIRCRAFT

RANGE,		DIRECT OPERATING COST,[2]	
STAT. MILES	(KM)	¢/ASSM	(¢/ASKM)
50	(80)	5.68	(3.53)
100	(161)	3.74	(2.32)
200	(322)	2.77	(1.72)
300[1]	(483)	2.54	(1.58)
400[1]	(644)	2.41	(1.50)
500[1]	(805)	2.37	(1.47)

NOTE 1. ADDITIONAL FUEL CAPACITY INSTALLED

2. UTILIZATION 2500 B.H/YR, AIRFRAME COST $90/LB

FIGURE 4.6-2
FUEL ECONOMY VERSUS RANGE, 100-PASSENGER STOL AIRCRAFT



5. PERFORMANCE

The D313 STOL aircraft was analyzed for wing-lift requirements, field length, conversion speed, climb-rate, descent-rate, and cruise speed. Performance capability was measured against the requirements of the NASA study guidelines and also the Federal Aviation Administration's "Tentative Airworthiness Standards for Powered Lift Transport Category Aircraft," Reference 5-1.

5.1 WING LIFT AND DRAG CHARACTERISTICS

The STOL tilt rotor employs the 23% thick wing of the VTOL tilt rotor with the added requirement to develop maximum wing lift at the fuselage attitudes encountered during takeoff and landing. This requirement called for the addition of leading-edge slats to extend the maximum wing C_L to +20 degrees fuselage angle of attack. Iteration on the maximum wing C_L required, indicated that the landing phase was critical with a required wing lift coefficient of 3.0. Takeoff was less critical with a required wing lift coefficient of 2.78. These values were met by a Fowler flap of 29% chord and a 23% thick GAW-1 airfoil. Wing lift characteristics for takeoff, landing and cruise are shown in Figure 5.1-1 and are based on the test data of References 5-2 and 5-3. The wing thickness tested was 17% so that the lift data of Figure 5.1-1 are considered to be slightly conservative.

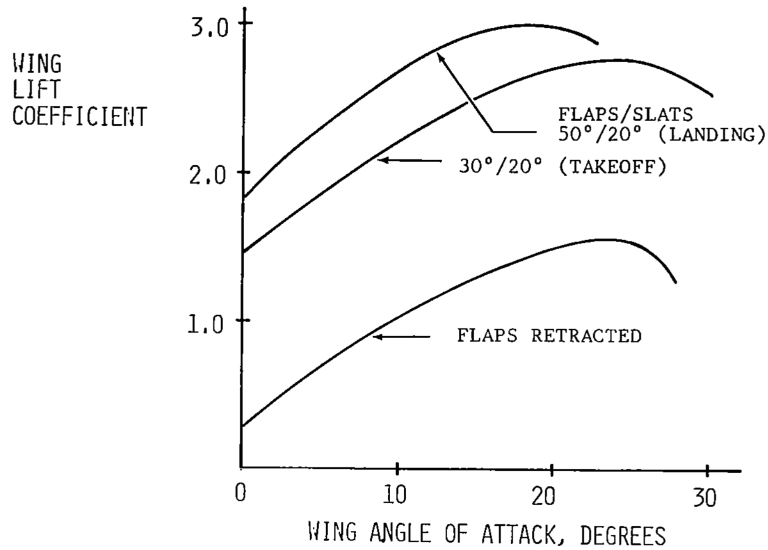
Wing drag characteristics for takeoff, landing and cruise are shown in Figure 5.1-1. These include an addition to the profile drag coefficients of References 5-2 and 5-3 to allow for the leading edge slat and the assumed 23% GAW-1 airfoil.

5.2 FIELD LENGTH

The NASA study guidelines specified a field length up to 2000 feet (610 m) with a clearance height of 35 feet (10.7 m) at either end. Ambient conditions were sea level 90°F (32.2°C). Initial study indicated that mission payload increased with field length so that a field length of 2000 feet (610 m) was selected. The ground rules used for determining takeoff and landing distances are shown in Table 5.2-1. These include those of the NASA study guidelines, the FAA Part XX, Reference 5-1, and those by this contractor.

5.2.1 TAKEOFF DISTANCE - Takeoff distances are shown in Table 5.2-2. The 2000/35-foot (610/10.7-meter) field requirement is met with adequate reserves. Takeoff is not critical due to the interconnected rotors (no asymmetric thrust) and the emergency power available from the engines. For instance, the four-engine takeoff power available at sea level 90°F

FIGURE 5.1-1
WING LIFT AND DRAG CHARACTERISTICS



GAW-1 AIRFOIL

$t/_c$ = 23%

ASPECT RATIO = 10

FOWLER FLAP 29% CHORD

L.E. SLAT 10% CHORD

REYNOLDS NO. 9 x 10^6

REF: NASA CR-2443
NASA TND-7428

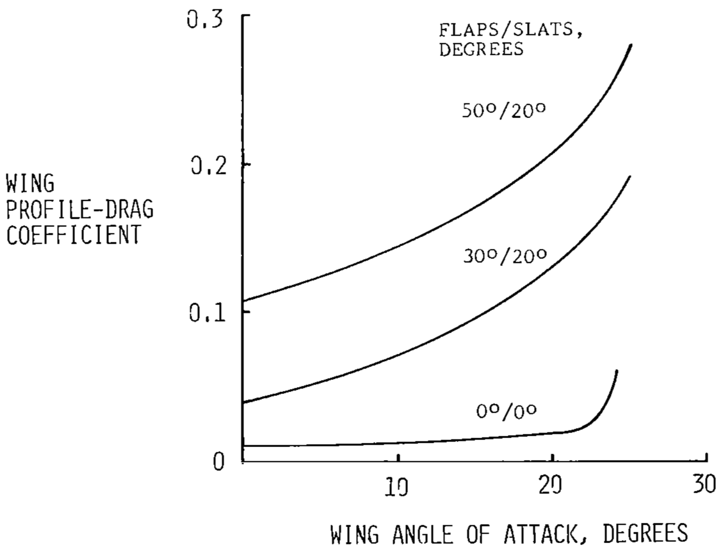


TABLE 5.2-1
TAKEOFF AND LANDING GROUND RULES FOR STOL TILT ROTOR AIRCRAFT (4 ENGINES)
SEA LEVEL $90^{\circ}F$

TAKEOFF	LANDING
<u>Acceleration</u>: Rolling friction coefficient = 0.03 All engines operating, 0.4g maximum acceleration. <u>Liftoff speed</u>: $V_{LOF} \geq 1.05(V_{MCA}$ and $V_{MCG})$ <u>Rotation</u>: 10 deg/sec maximum <u>Climbout conditions to 35-ft obstacle</u>: AEO: Climb ≥ 6.7%(15:1)gradient,250 fpm (gear down) OE1: Climb ≥ 6.7%(15:1)gradient,250fpm (gear up) α AT C_L MAX(T.0)-5° (gear down) <u>Speed at obstacle</u>: $V_2 \geq V_{LOF} \geq 1.10V_{MCA} \geq V_{MCA}+15$ kt <u>Factors for field length</u>: 1.15 for all engines operating 1.00 for engine cut at liftoff 1 second delay, pilot reaction 1 second delay to increase power 1.00 for accelerate-stop 1 second delay at V_1, pilot reaction 1 second delay to apply brakes	<u>Approach speed</u>: (Speed at 35-ft obstacle) $V_{AP} \geq 1.10V_{MCA} \geq V_{MCA}$ + 15 kt α AT C_L MAX (APPROACH)-10°(gear down) <u>Landing climbout</u>: AEO: Climb ≥ 3.33%(30:1)gradient, 250 fpm (gear down) OE1: Climb ≥ 3.33%(30:1)gradient, 250 fpm (gear up) <u>Flight path from 35 ft</u>: Maximum rate of descent At 35 ft: 800 fpm At touchdown: 300 fpm <u>Rotation</u>: 10 deg/sec maximum <u>Deceleration</u>: 1 sec time delay Braking friction coefficient = 0.35 Maximum deceleration on ground = 0.4g <u>Factor for field length</u>: Landing distance from 35 ft divided by 0.75

AEO = All engines operating
OE1 = One engine inoperative

TABLE 5.2-2
TAKEOFF DISTANCE REQUIRED, 100-PASSENGER STOL AIRCRAFT.
DESIGN FIELD: 2000 FT, CLEARANCE HEIGHT = 35 FT

TAKEOFF AT:

- DESIGN GROSS WEIGHT
- SEA LEVEL $90^{\circ}F$
- V_1 = 65 KNOT, V_2 = 80 KNOT
- MAST ANGLE = 60°
- FLAPS $30^{\circ}/20^{\circ}$

	DISTANCE TO 35-FEET ALTITUDE
ALL ENGINES OPERATING (AEO)	1605 FEET (489.2 M)
1.15 X AEO DISTANCE	1846 FEET (562.7 M)
ONE ENGINE CUT AT V_1	1651 FEET (503.2 M)
	DISTANCE TO STOP
ACCELERATE TO V_1 AND STOP (2-SECOND DELAY AT V_1)	1282 FEET (390.8 M)

(32.2°C) is 7763 shp (5789 kw). The transmission limit is 6977 shp (5203 kw). If an engine fails, the power to the rotors drops from 6977 shp (5203 kw) to 5233 shp (3902 kw) for one second, the pilot selects emergency power (2½-minute rating) and one second later (engine response time) the power available is back up to the transmission limit of 6977 shp (5203 kw). This increased the takeoff distance by only 46 feet (14.0 m).

5.2.2 LANDING DISTANCE - Landing distances are shown in Figure 5.2-3. With a field length factor of 0.75 the aircraft is required to stop in 1500 feet (457 m) from the threshold. The distance required is 1488 feet (454 m) with a dry runway and the allowable deceleration of 0.4g. Reverse rotor thrust is available (this could determine lower collective pitch limit) and at a mast angle of 60° the average deceleration from aerodynamic braking alone was calculated to be 0.32g. The total distance required to stop was then 1670 feet (509 m). This could be reduced to 1500 feet (457 m) if the pylons were converted from 60° to 45° (time required 1.5 seconds) during the landing roll. The control system could command this automatically upon selection of reverse rotor thrust.

Thus, the landing distance is the critical performance parameter and for a given rotor system sets an upper limit on gross weight.

5.3 CONVERSION CORRIDOR

The conversion corridor (Figure 5.3-1) is slightly wider than that of the VTOL tilt rotor because of the high-lift flap system, and exceeds 90 knots (167 kph) in "width."

A typical sequence at takeoff would be:

- climb-out at 80 knots (148 kph) airspeed to 1000 feet (305 m) altitude
- accelerate to 100 knots (185 kph)
- convert pylons to airplane mode
- accelerate to 140 knots (259 kph)
- retract flaps

If desired, the pylons can be converted in climbing-, level-, or descending-flight.

TABLE 5.2-3
LANDING DISTANCE REQUIRED, 100-PASSENGER STOL AIRCRAFT.
DESIGN FIELD: 2000 FEET, CLEARANCE HEIGHT 35 FEET

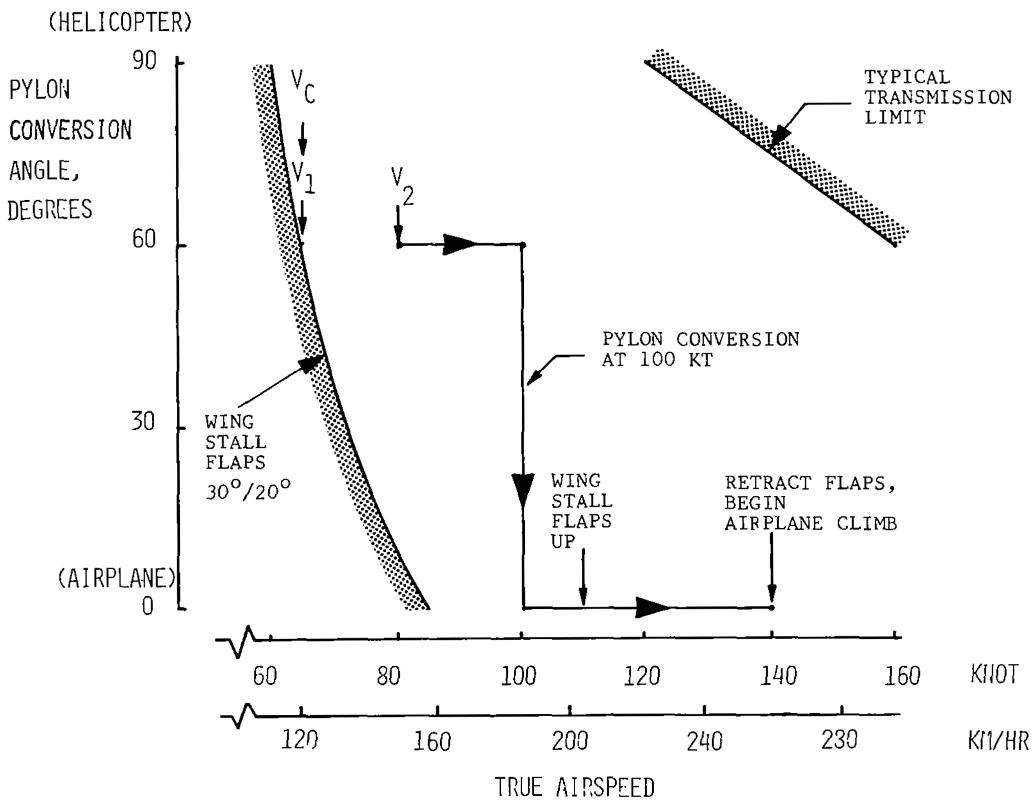
APPROACH:

- DESIGN GROSS WEIGHT
- SEA LEVEL $90^{\circ}F$
- 80 KNOT
- DESCENT RATE 800 FEET/MINUTE
- MAST ANGLE 60°

DISTANCE TO STOP WITH BRAKING (ALLOWABLE DECELERATION = 0.4G)	1488 FEET	(453.5 M)
DISTANCE TO STOP WITH REVERSE ROTOR THRUST ONLY	1670 FEET[1]	(509.0 M)

NOTE 1. CAN BE REDUCED BY MAST CONVERSION FROM 60° TO 45° DURING LANDING ROLL

FIGURE 5.3-1
CONVERSION CORRIDOR, 100-PASSENGER STOL AIRCRAFT.



5.4 CLIMB PERFORMANCE

Climb performance was calculated using Bell program IFHB 75. This program includes analytical methods to calculate the power required by a tilt rotor configuration in all flight modes. The analytical methods have been correlated with BHC wind tunnel test data and are currently being used for the XV-15. The program was modified to allow for the wing characteristics of the STOL tilt rotor.

Wind tunnel test data have shown that at speeds above 40 knots (74 kph) and mast angles less than 60 degrees, the rotor interference on the wing is essentially zero. This contributed significantly to the high lifting efficiency of the D313.

5.4.1 RATE-OF-CLIMB IN TAKEOFF CONFIGURATION - The climb capability of the D313 in takeoff configuration, pylons 60°, flaps 30°/20°, at sea level 90°F (32.2°C), is shown in Figure 5.4-1. At V_2 (80 knots, 148 kph), the climb rate is 1520 ft/min (463 m/min) at the transmission limit (90% IRP). This exceeds the required climb rate, 540 ft/min (165 m/min), of the study guidelines. With one engine out the climb rate is 1060 ft/min (323 m/min) at the IRP rating (30-minute) of the remaining engines, and 1520 ft/min (463 m/min) at the 2½-minute rating. The required climb rate in FAA Part XX on three engines is 250 ft/min (76.2 m/min). Note that the wing stall speed reduces with climb rate. This is due to the rotors carrying an increasing portion of gross weight as climb rate is increased. Thus, when climbing at the transmission limit there is a margin of 25 knots (46 kph) between wing stall (55 knots, 102 kph) and V_2 (80 knots, 148 kph). Wing stall speed in level flight is 64 knots (119 kph).

5.4.2 RATE-OF-CLIMB IN AIRPLANE CONFIGURATION, FLAPS 30°/20°
The climb capability of the D313 in airplane configuration, pylons 0°, flaps 30°/20°, is shown at sea level 90°F, in Figure 5.4.2. Rotor tip speed is 600 ft/sec (183 m/sec) and the transmission limit is reduced with tipspeed to 5980 shp (4459 kw). The climb rate at a typical conversion speed of 100 knots (185 kph) is 1240 ft/min (378 m/min). Wing stall speed is 85 knots (157 kph) in level flight.

5.4.3 RATE-OF-CLIMB IN AIRPLANE CONFIGURATION, FLAPS RETRACTED
The climb capability of the D313 in airplane configuration, pylons 0°, flaps retracted, is shown at sea level 90°F, in Figure 5.4-3. The maximum climb rate is 1520 ft/min (463 m/min) at 140 knots (259 kph) at the transmission limit. The FAA Part XX requires a 1.7% climb gradient (417 ft/min (127 m/min) at 140 knots) with 3 engines at maximum continuous power

FIGURE 5.4-1
RATE OF CLIMB IN TAKE-OFF CONFIGURATION,
100-PASSENGER STOL AIRCRAFT.

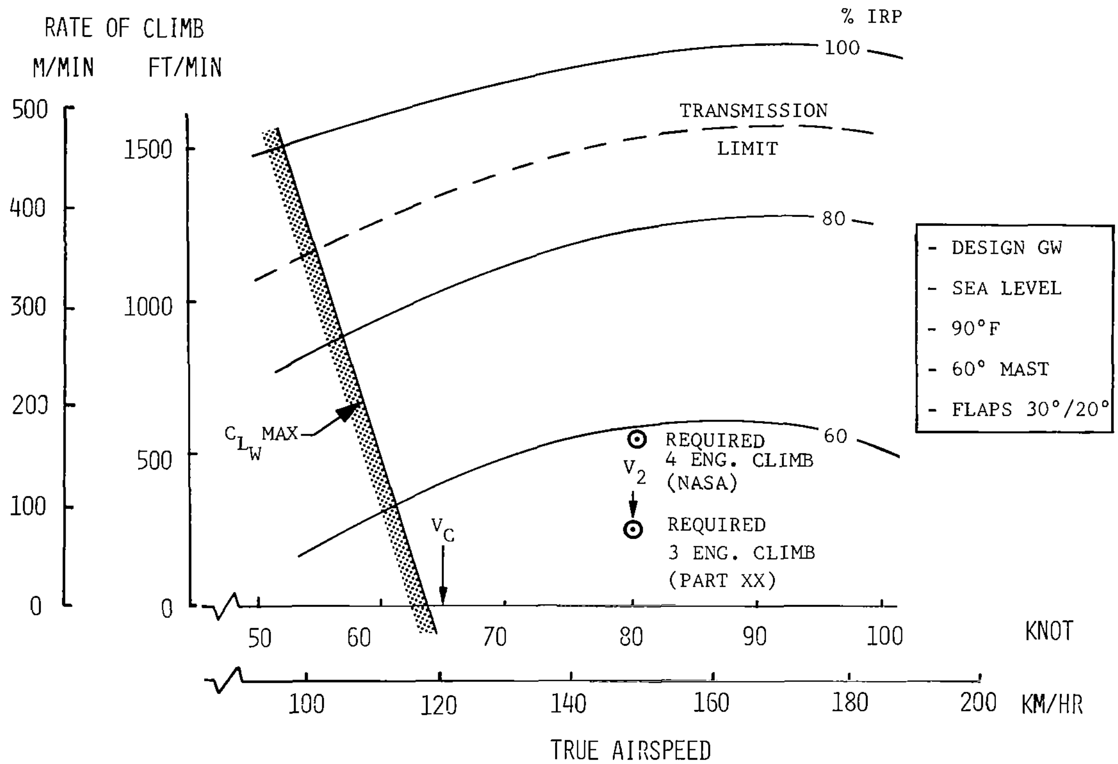


FIGURE 5.4-2
RATE OF CLIMB IN AIRPLANE CONFIGURATION, FLAPS ($30^{\circ}/20^{\circ}$), 100-PASSENGER STOL AIRCRAFT.

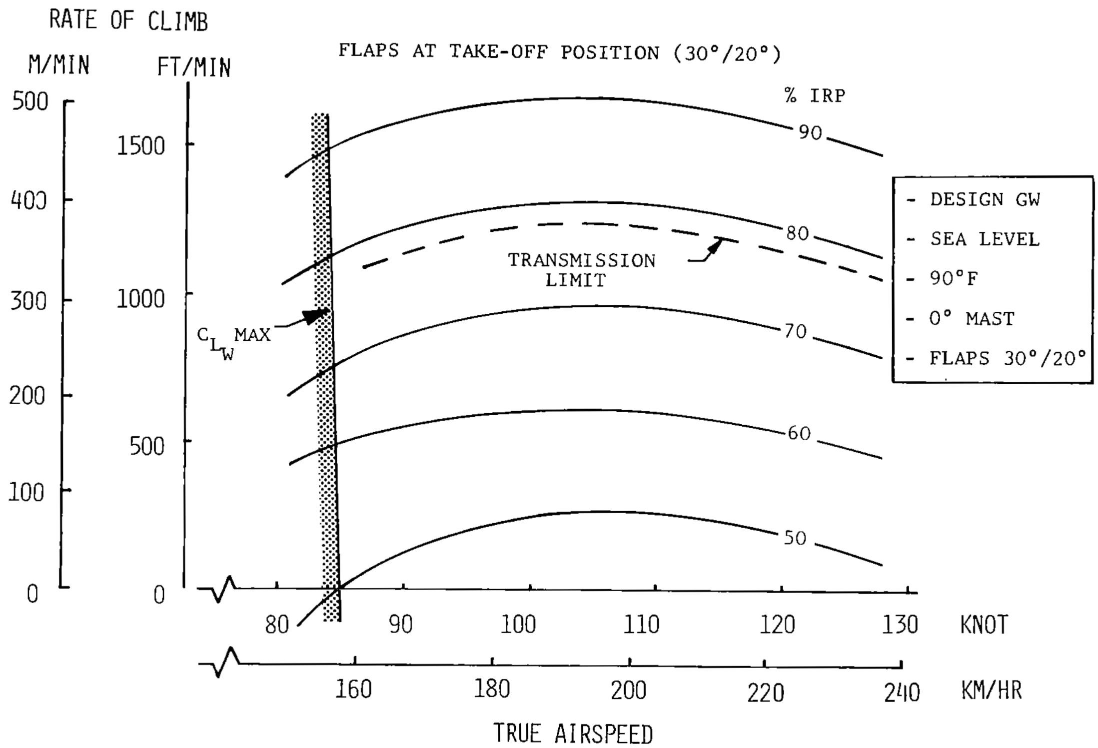
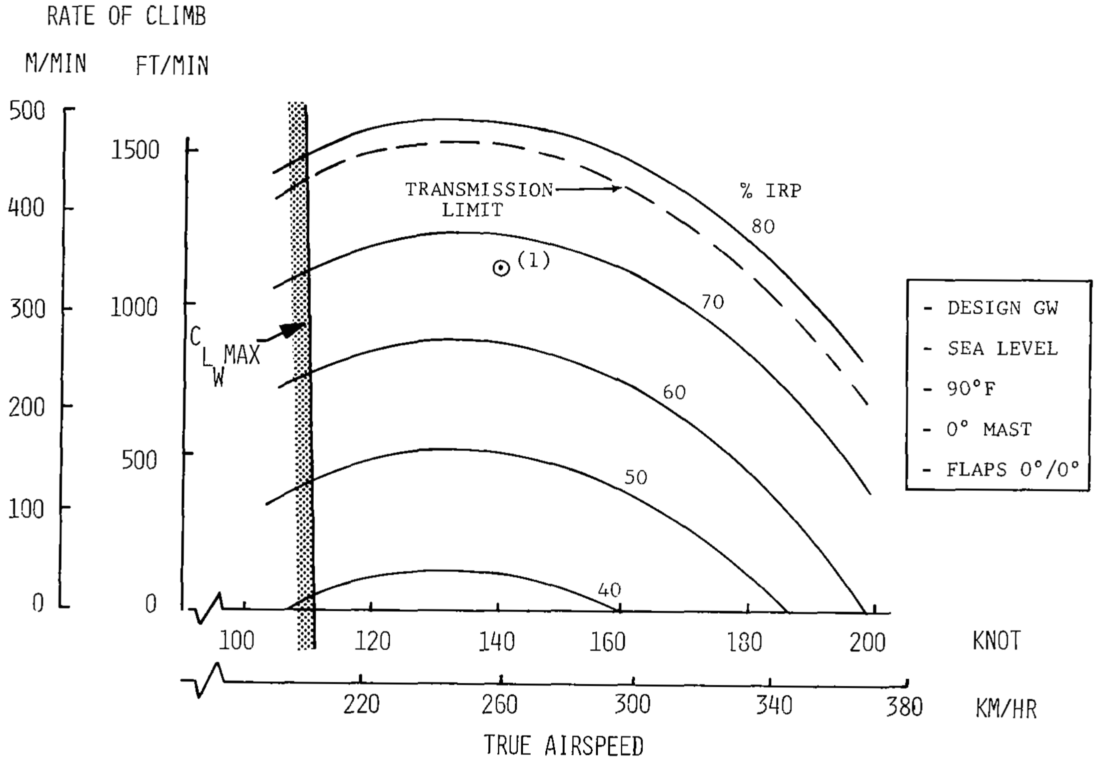


FIGURE 5.4-3
RATE OF CLIMB IN AIRPLANE CONFIGURATION, FLAPS RETRACTED,
100-PASSENGER STOL AIRCRAFT



NOTE 1. RATE OF CLIMB = 1140 FEET/MIN WITH 3 ENGINES AT MAX. CONTINUOUS POWER. FAA PART XX REQUIRES 417 FEET/MIN. (1.7% GRADIENT)

(68.2% IRP). The D313 climb rate is 1140 ft/min (347 m/min) for the above conditions. Wing stall speed in level flight is 111 knots (206 kph).

5.4.4 RATE OF CLIMB IN LANDING CONFIGURATION, FLAPS 50°/20° - The climb and descent capability of the D313 in landing configuration, pylons 60°, flaps 50°/20°, is shown at sea level 90°F (32.2°C) in Figure 5.4-4. A typical approach at 80 knots (148 kph), descent rate 800 ft/min (244 m/min) is shown to require 28% IRP. Fuselage attitude was calculated to be +3.5°. For a baulked landing or "landing climbout" a positive climb rate on 4 engines at minimum flying speed (V_{LOF} = 68 kts, 126 kph) is required by:

- NASA study guidelines, 3.33% gradient (230 ft/min, 70.1 m/min) required

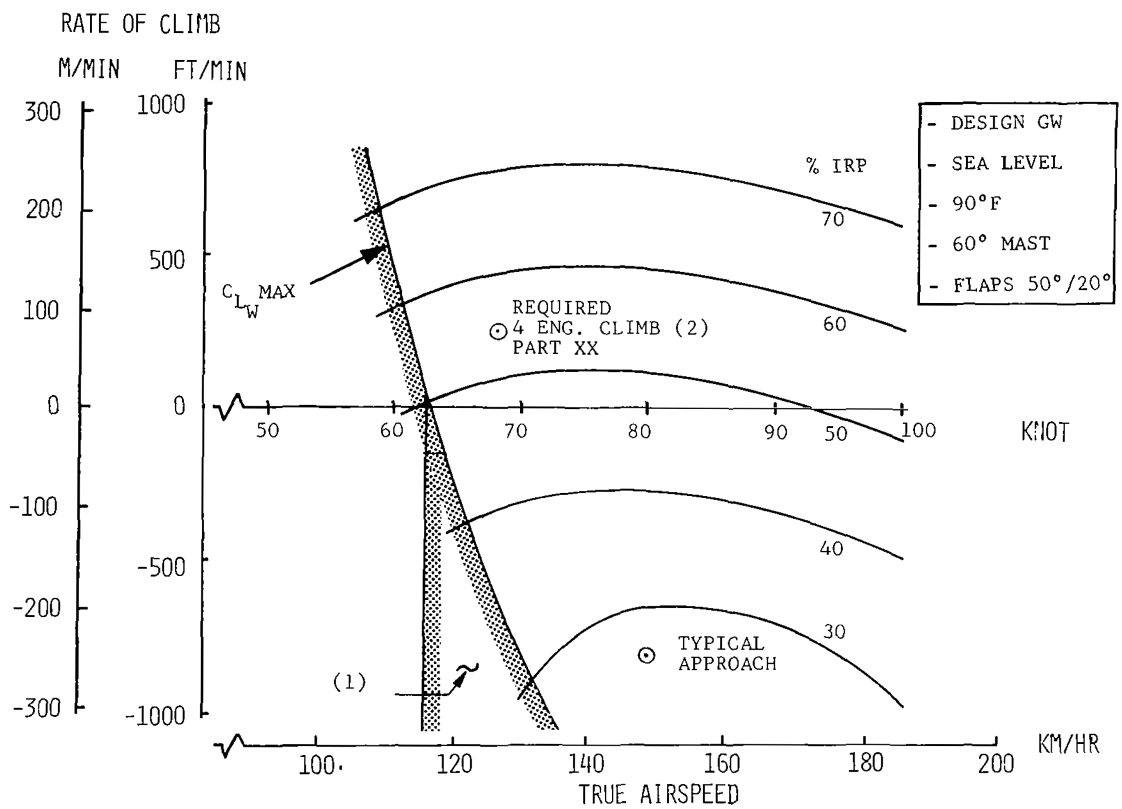
- FAA, Part XX, 250 ft/min (76.2 m/min) required

The D313 has a climb rate exceeding 1000 ft/min (305 m/min) in these conditions and thus exceeds requirements. Wing stall in level flight is 62 knots (115 kph).

At the approach condition shown (80 knots (140 kph), 800 ft/min (244 m/min) descent rate) the speed margin from maximum wing lift is 9 knots and the angle of attack margin is 8 degrees. These are less than desirable and this condition required the addition of the full-span wing spoilers. A brief description of spoiler characteristics follows. As the pilot begins to descend on final approach, he would raise the spoilers to reduce wing lift approximately 40% and increase drag. The rotor thrust is increased to carry the dumped-lift. Because the spoiled wing has a 10-degree increase in angle of attack at its spoiled maximum value, Reference 5-2, the approach condition with spoilers-up is predicted to have adequate margins. Detailed wind tunnel testing of this feature is required. The landing flare (typically 1.23 g) can be made on collective pitch alone or the spoilers could be linked to collective pitch (only in the takeoff and landing configurations) such that the spoilers close as collective pitch is raised. Closing the spoilers fully would produce a 1.31 g flare.

It should be noted that the spoilers enable the rotor thrust vectors to be increased from 7000 lbf/rotor (28.3 kN) to 17000 lbf/rotor (68.8 kN) (typically) in the final approach descent. This enables significant control characteristics to be achieved in crosswinds as discussed next.

FIGURE 5.4-4
RATE OF CLIMB IN LANDING CONFIGURATION, FLAPS (50°/20°),
100-PASSENGER STOL AIRCRAFT.



NOTE 1. SPOILERS DUMP EXCESSIVE WING LIFT DURING DESCENT AND INCREASE WING ANGLE-OF-ATTACK MARGIN.

2. RATE OF CLIMB CAPABILITY = 1220 FT/MIN @ 90% IRP (TRANSMISSION LIMIT).
RATE OF CLIMB REQUIRED = 250 FT/MIN (FAA PART XX), 230 FT/MIN (NASA).

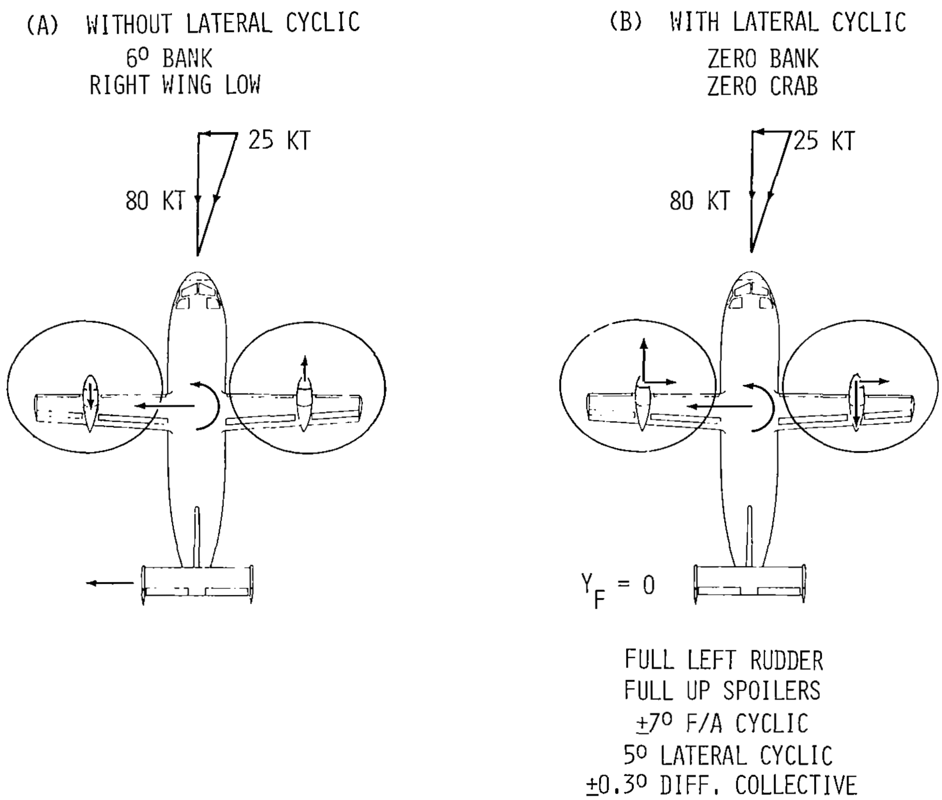
5.4.5 CONTROL CHARACTERISTICS IN CROSSWINDS - The crosswind approach and flare is a demanding piloting task. Present typical CTOL airline operations (B-727) with 120-140 knot approach speeds have set a 25-knot limit on the crosswind component. The NASA guidelines for this study included a 25-knot design crosswind, but at the lower STOL approach speeds (80 knot for D313) the larger crosswind crab and bank angles can significantly add to the piloting task.

Two crosswind approaches for the D313 are shown in Figure 5.4-5, with the fuselage aligned with the runway center line (zero crab - angle). The aircraft is in the landing configuration with pylons 60°, flaps $50^{\circ}/20^{\circ}$ and is approaching at 80 knots and a descent rate of 800 ft/min (244 m/min). The 25-knot crosswind is from the right.

Case (a) shows a typical trimmed approach for the D313 without lateral cyclic rotor pitch. The fin tends to weathervane the nose to the right. Left rudder is applied to hold the nose on the runway centerline; however, left rudder also applies nose-left differential cyclic which adds to the nose-left fuselage moment. The nose-left moments are balanced by the fin nose-right moment which is generated by the fin side-force. To balance the fin and fuselage side forces to the left, the pilot holds the right wing low at a 6° bank. This is disliked by the passengers and also adds to the piloting task during the flare.

Case (b) shows a zero-bank zero-crab approach for the D313 with lateral cyclic rotor pitch. The full span spoilers have been raised to increase the rotor force-vectors and rotor control power. Right lateral cyclic (5-degrees or 42 percent of available travel), commanded by a trim wheel setting, balances the fuselage side force and application of left rudder holds the net fin force to zero. The nose-left fuselage moment is balanced by 7-degrees of differential longitudinal cyclic also commanded by the same trim wheel. The right lateral cyclic produces a roll-right moment which is balanced by ±0.3 degrees of differential collective (±3.0 degrees available) commanded by moving the stick slightly to the left. The total cyclic pitch required is 8.6 degrees (square root of the sum of the squares of lateral and longitudinal cyclic) which is 72 percent of that available. The flare on landing can utilize collective pitch or the wing spoilers. At this preliminary stage, the spoilers seem more attractive since retraction of the spoilers would add a near-vertical lift vector which would not disturb the trimmed side-force and yaw-moment conditions. The wing spoilers obviously require the same reliability as the rotor control system.

FIGURE 5.4-5
CROSSWIND APPROACH CAPABILITY, STOL TILT ROTOR AIRCRAFT.
FLAPS ($50^{o}/20^{o}$), R/D 800 FEET/MINUTE



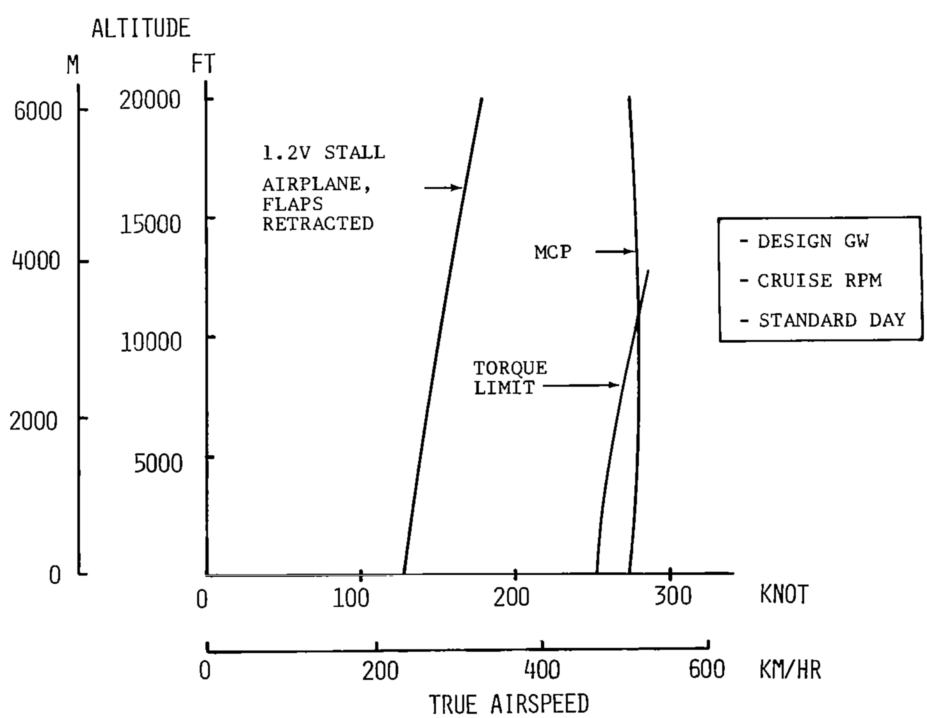
The piloting task in the critical crosswind approach has been simplified because of the unique thrust vector-tilt control capabilities of the tilt rotor system. This will result in fewer delays in severe crosswinds and in less fatigue damage to the landing gear.

The definition of side-force control capabilities available with lateral cyclic trim is an excellent advanced research task with the XV-15 aircraft.

5.5 CRUISE PERFORMANCE

The airplane cruise envelope for the D313, with all engines operating, is shown in Figure 5.5-1. The lower limit is at 1.2x wing stall (based on a maximum wing lift coefficient of 1.58, flaps retracted). The upper boundary is limited by maximum continuous power (MCP) or by the torque limit of the drive system. Cruise speed at MCP is 278 knots (515 kph) at 11000 feet (3353 m) and 274 knots (507 kph) at 20000 feet (6096 m).

FIGURE 5.5-1
CRUISE PERFORMANCE, ALL ENGINES OPERATING,
100-PASSENGER STOL AIRCRAFT.



6. NOISE CHARACTERISTICS

Tilt rotor noise levels are calculated with the BHC rotorcraft noise prediction computer program KA9701. This procedure uses the analytical formulation of Lowson and Ollerhead (Reference 6-1) and also correlation with experimental data. For this study whirl test data of the BHC Model 300 tilt rotor at Wright-Patterson Air Force Base (Reference 6-2) were used for correlation. This rotor is identical to the right-hand rotor of the XV-15.

6.1 NOISE CONTOURS AT TAKEOFF

The D313 noise contours for takeoff at sea level 90°F are shown in Figure 6.1-1. Climb rate is 1400 ft/min (427 m/min) at 80 knots (148 kph). Climb gradient is +9.9 degrees and the fuselage pitch attitude is +10.2 degrees. The perceived noise at the 500 foot (152 m) sideline is 96 PNdB. The area within the 95 PNdB contour is 113.4 acres (.459 sq km).

6.2 NOISE CONTOURS AT LANDING

The D313 noise contours for landing at sea level 90°F are shown in Figure 6.1-2. Descent rate is 800 ft/min (244 m/min) (NASA specified maximum) at 80 knots (148 kph). Approach gradient is -5.6 degrees and the fuselage pitch attitude is +3.5 degrees. The perceived noise at the 500 foot (152 m) sideline is 90 PNdB. The area within the 95 PNdB contour is 57.9 acres (.234 sq km).

FIGURE 6.1-1
NOISE FOOTPRINT, TAKEOFF, 100-PASSENGER STOL AIRCRAFT

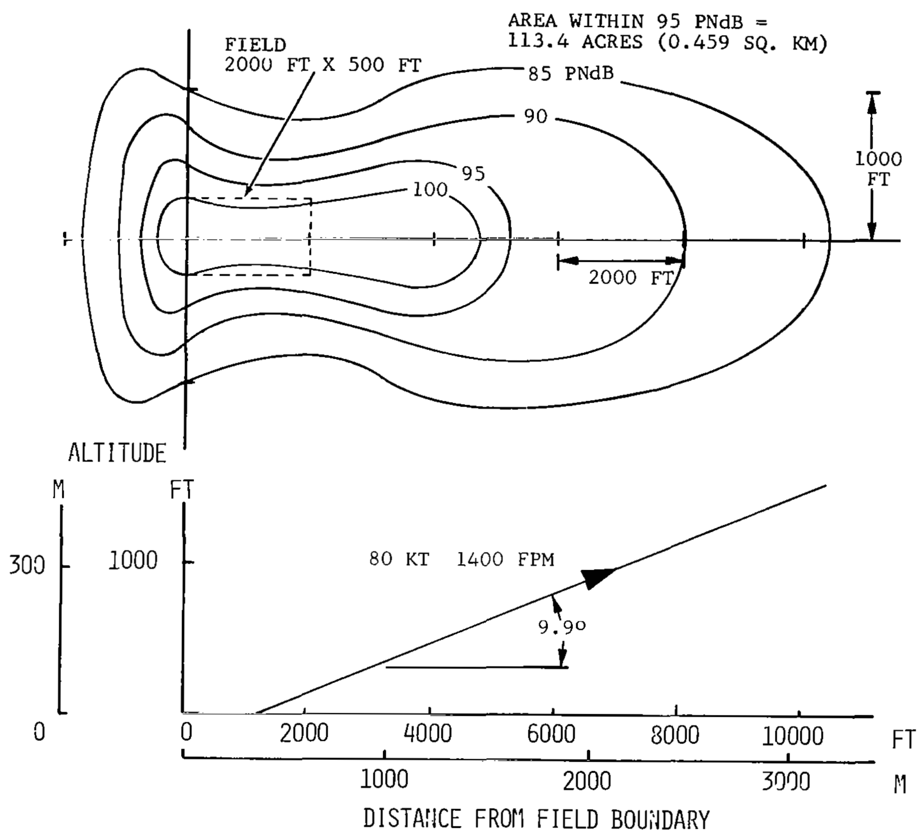
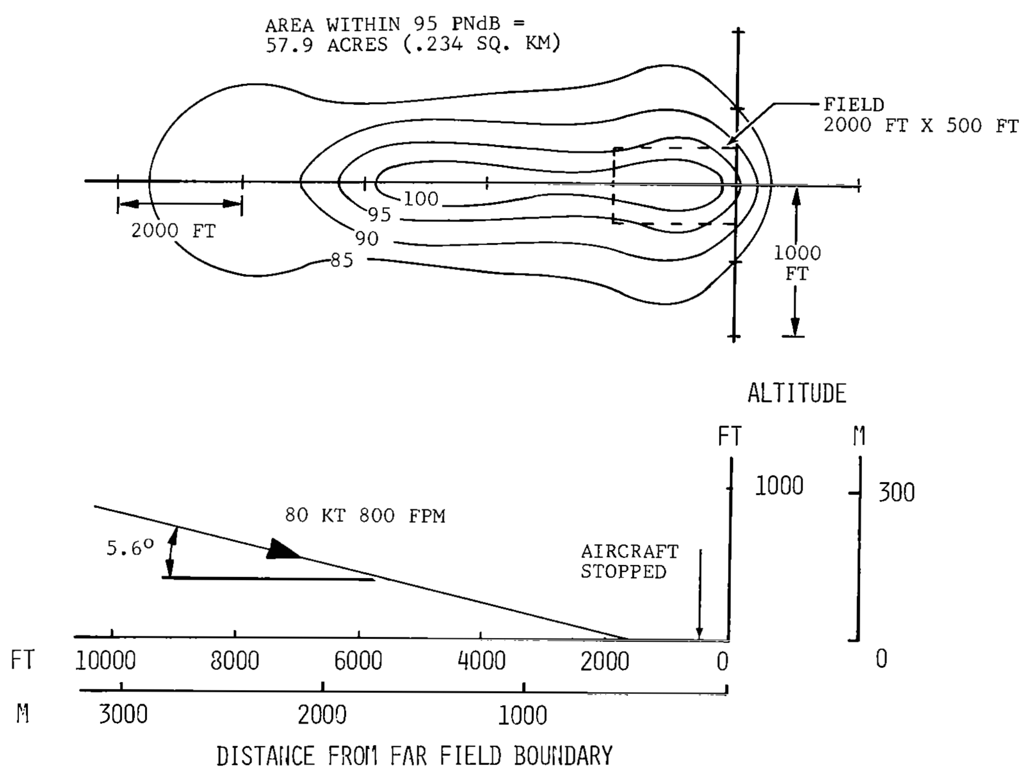


FIGURE 6.1-2
NOISE FOOTPRINT, LANDING, 100-PASSENGER STOL AIRCRAFT



7. HANDLING QUALITIES

The stability, control and handling qualities analyses of the D313 100-passenger STOL point design are based on the results obtained from a digital version of the NASA tilt rotor flight simulation computer program. This program is described below. Definition of the configuration, inputs for the program, and the relationships to the XV-15 are described in Section 7.2. The following handling qualities topics are discussed in the subsequent paragraphs: static trim stability, dynamic stability, control power and attitude response, cruise flight maneuver stability and several conclusions. Low speed gust response is discussed in Section 9 "SAFETY ASPECTS."

7.1 BASIS FOR ANALYSIS

The stability, control and handling qualities analysis is based on results obtained from a digital version of the NASA tilt rotor flight simulation program, designated BHC Program IFHB75. The math model includes a six-degree-of-freedom trim iteration routine which provides the capability to analyze lateral/directional characteristics, including the effects of a steady-state crosswind condition throughout the flight envelope. Gust and control response predictions are included in the dynamic phase of the model; however, inputs are currently limited to step functions for both cases. The program now includes the capability of evaluating dynamic stability roots with the XV-15 Stability and Control Augmentation System (SCAS) operating in order to predict the improvements over those of the basic aircraft. Improved thrust and horsepower correlation with full-scale test data recently have been incorporated in the math model.

7.2 CONFIGURATION DEFINITION

The 100-passenger STOL tilt rotor configuration analyzed for this study possesses certain identical characteristics to those of the XV-15 aircraft, as follows: blade section properties (i.e., twist, lift and drag coefficients, precone angle and tip loss factor); rotor-on-empennage and wing-on-empennage induced flow characteristics; cockpit control travels, rotor cyclic and collective riggings (with the exceptions noted below); elevator rigging; and rotor/engine governor characteristics.

Portions of the STOL configuration which are independent or different from that of the XV-15 were evaluated and incorporated separately into the math model. These were:

- Fuselage pitching moment variation with angle of attack ($M\alpha$) and yawing moment variation with sideslip angle ($N\beta$). These changed the required empennage volume coefficients.

- Wing-flap lift, drag and pitching moment coefficients.

- Wing lateral/directional derivatives.

- Spoiler lift, roll and yaw coefficients and rigging.

- Rudder chord increased from 25% (XV-15) to 40% of fin chord. Rudder travel increased from ±20° to ±25°.

- Differential collective pitch is rigged to the pedals as well as the lateral stick, to increase yaw control power.

- Differential F/A cyclic rigging. However, approximately the same differential cyclic travel (with pedal) is available during an STOL takeoff or landing (mast angle = 60°, 80 knot airspeed) as the XV-15 in hover with the mast vertical.

- Horizontal stabilizer incidence 4° nose down, to reduce the downwind rotor flapping in a crosswind approach.

Throughout the analyses, design gross weight was used and the center-of-gravity range is that defined in paragraph 4.7 of the Design Criteria; i.e., payload shift of ±5 percent of the passenger cabin length. The basic geometric data (rotor, fuselage, wing/pylon, landing gear sizes and locations), weight, center-of-gravity, rotor rpm and scaled parameters (such as the blade dynamic characteristics, engine rated power, and total aircraft inertias) were defined by the design synthesis method.

7.3 STATIC TRIM STABILITY

Longitudinal control position and aircraft pitch attitude for trimmed level flight throughout the speed and conversion angle ranges are shown in Figure 7.3-1 for both the forward and aft center-of-gravity locations. Because only small differences exist in control position with flaps/slats in takeoff (30°/20°) and landing (50°/20°) positions, only the landing configuration is shown. The takeoff, conversion and landing data represent sea-level tropical day conditions with a 25-knot crosswind and helicopter mode rpm whereas the airplane configuration represents cruise conditions at 20000 feet, standard day without a crosswind and at the lower cruise tipspeed. A fly-by-wire control system or a slightly larger horizontal tail area would improve the shallow stick gradient in the airplane mode.

Static trim stability for the following climb and descent conditions, with a 25-knot crosswind, is shown in Figure 7.3-2: takeoff climb (1400 fpm, 427 m/min), landing climb (260 fpm, 79 m/min) and landing descent (800 fpm, 244 m/min). The

FIGURE 7.3-1
STATIC TRIM STABILITY, LEVEL FLIGHT
100-PASSENGER STOL AIRCRAFT.
DGW = 64300 LBF

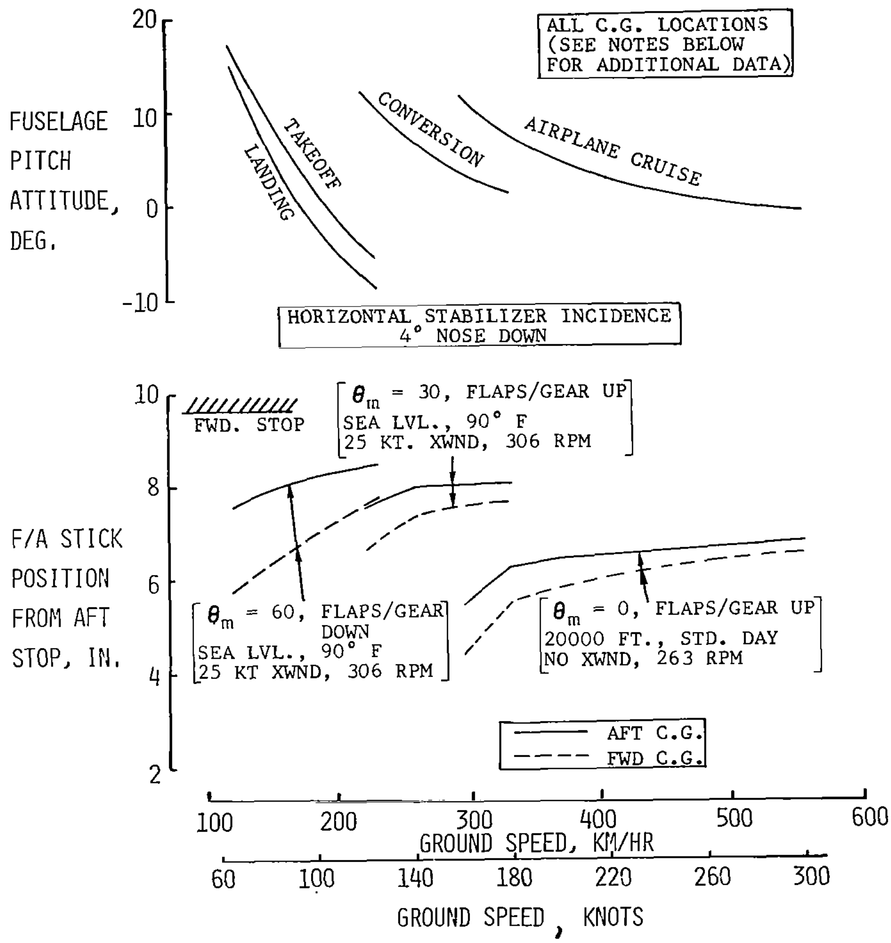
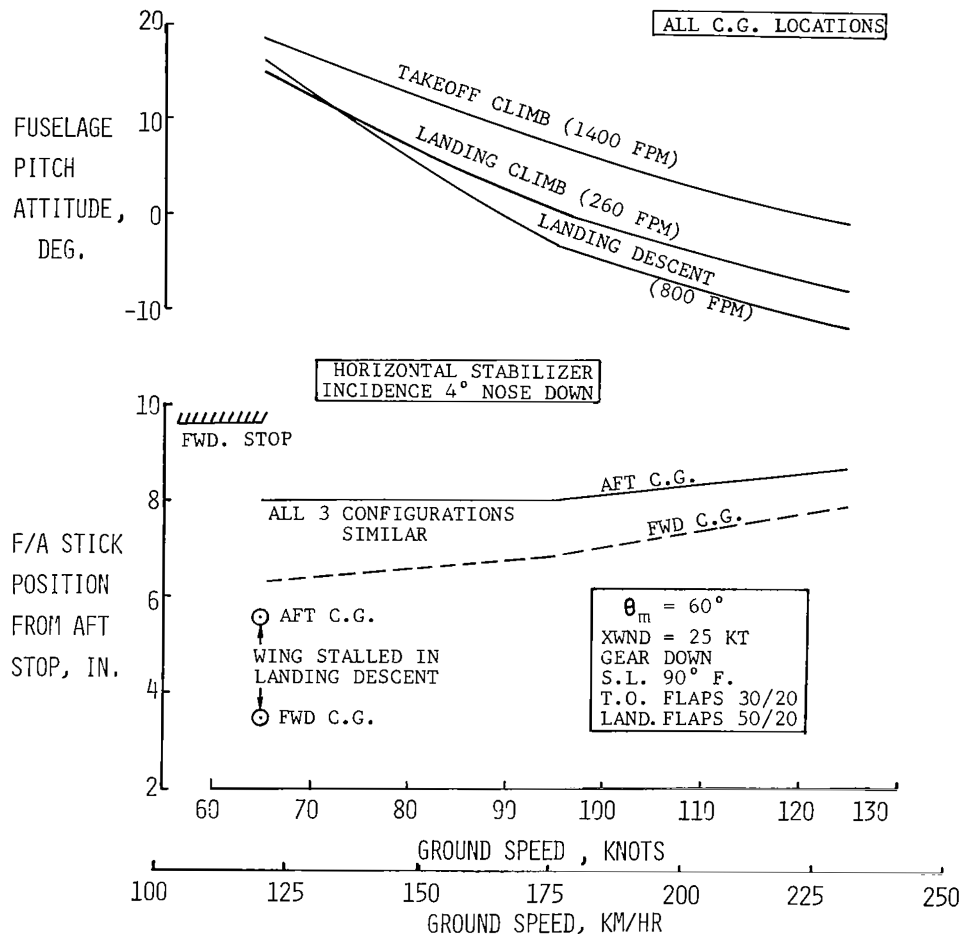


FIGURE 7.3-2
STATIC TRIM STABILITY, CLIMB AND DESCENT,
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



longitudinal control positions for each flight condition are similar since collective pitch is the primary variable; therefore, only one F/A stick value is presented for each speed. Although the aircraft is still in a trimmed flight condition with sufficient rotor thrust margin available, the wing is beyond stall during a landing descent at the minimum control speed (with spoilers retracted) and thereby results in a more aft control position and higher fuselage pitch attitude than would occur with an unstalled condition. This condition is rectified by the wing spoilers as discussed in Section 5.4.4. The fuselage pitch attitudes are all within the specified limits of paragraph 5.1 of the Design Criteria (+20°, -10°) with the exception of the landing descent configuration above 117 knots (217 kph) which is well beyond the mission profile.

7.4 DYNAMIC STABILITY

Level flight dynamic stability for three oscillatory flight modes is presented for the takeoff configuration in Figures 7.4-1 and 7.4-2, for the airplane cruise configuration in Figures 7.4-3 and 7.4-4, and for the landing configuration in Figures 7.4-5 and 7.4-6. The longitudinal Short Period mode is shown on the first figure of each configuration set and the longitudinal Phugoid mode and lateral/directional Dutch Roll mode are shown on the second figure of each configuration set. The Roll and Spiral modes are stable throughout the flight envelope shown. Both the aft and forward center-of-gravity roots are indicated on each figure. For those configurations where the basic aircraft stability roots (SCAS-off) are not within the Level 1 limits of the Design Criteria (paragraph 1.1.4), SCAS-on roots are shown using the XV-15 SCAS (gains, time constants, etc.)

In the takeoff configuration, the short period mode (Figure 7.4-1) at forward c.g. meets Level 1 above 95 knots (176 kph) without SCAS and the SCAS brings this mode into the Level 1 region below this speed. At aft c.g., the short period mode becomes stable aperiodic without SCAS; however, the SCAS again brings this mode into the Level 1 region. For the Phugoid and Dutch Roll modes during a takeoff climb (Figure 7.4-2), SCAS is required to provide sufficient damping for the stable Level 1 region.

In the airplane cruise configuration, the short period mode (Figure 7.4-3) at forward c.g. meets Level 1 between 160 knots (297 kph) and 280 knots (519 kph) while the aft c.g. roots are aperiodic beyond 220 knots (408 kph). By reconfiguring the SCAS or increasing the horizontal stabilizer area by 13% (from 270 to 305 ft^2), the aft c.g. short period roots would fall into the oscillatory Level 1 region. Figure 7.4-4 indicates that both the Phugoid and the Dutch Roll modes of the basic

FIGURE 7.4-1
DYNAMIC STABILITY, TAKE-OFF CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

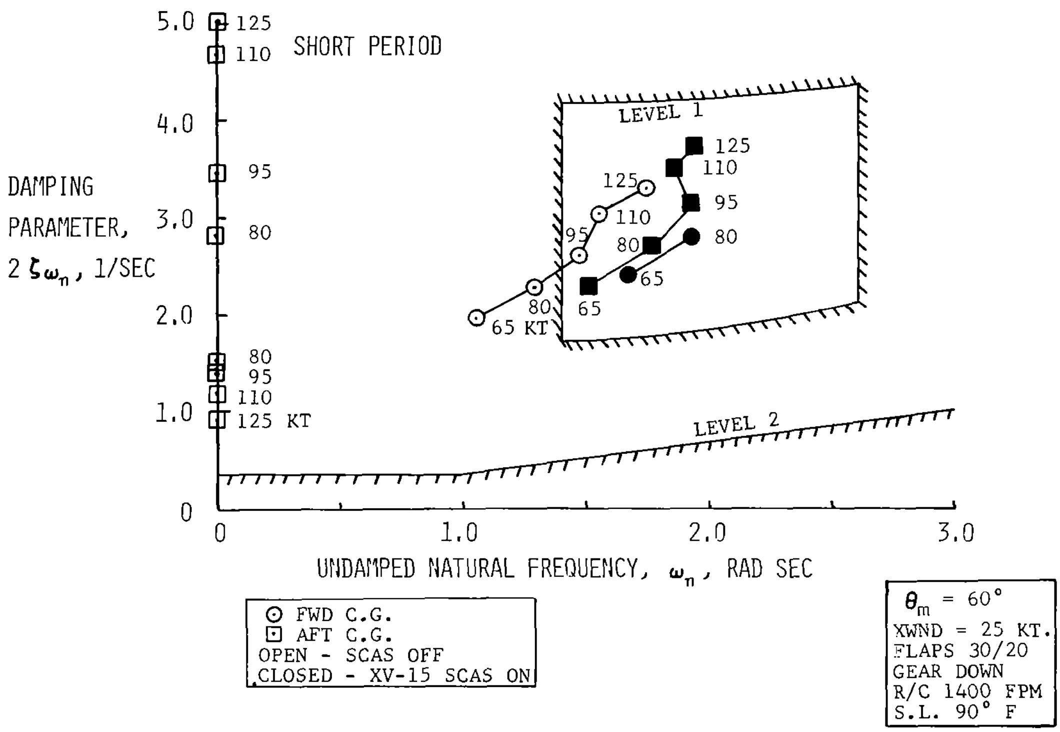


FIGURE 7.4-2
DYNAMIC STABILITY, TAKE-OFF CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

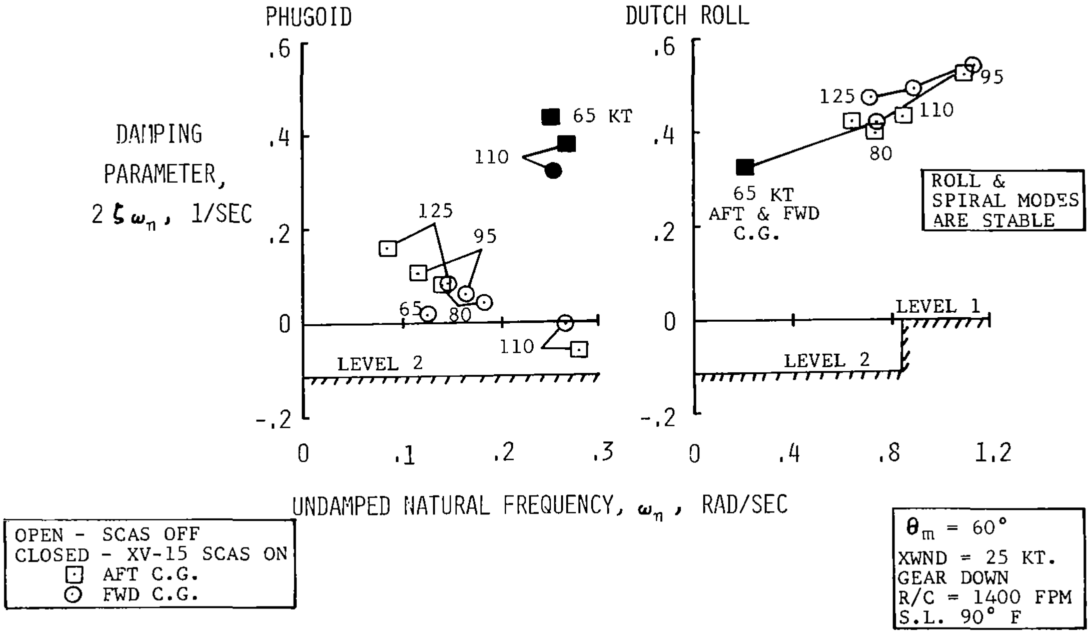


FIGURE 7.4-3
DYNAMIC STABILITY, AIRPLANE CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

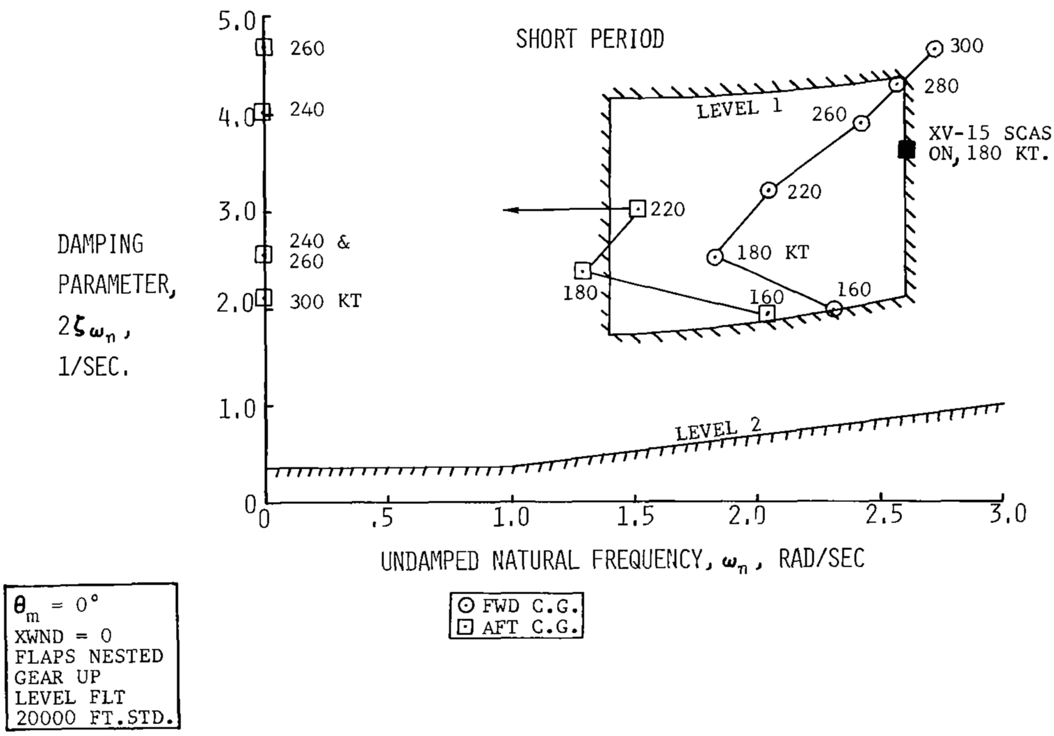


FIGURE 7.4-4
DYNAMIC STABILITY, AIRPLANE CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

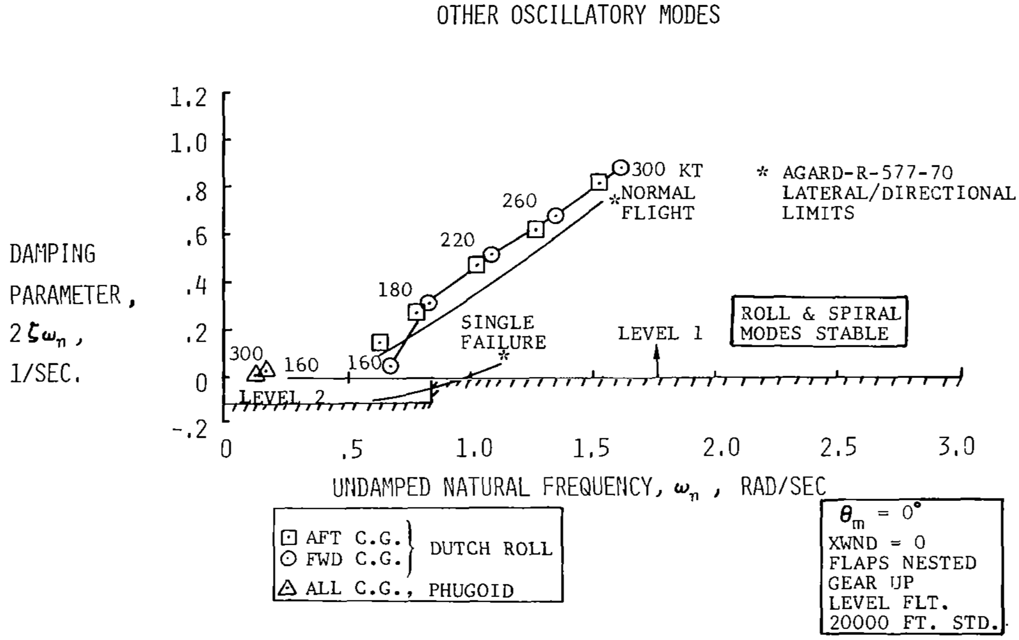


FIGURE 7.4-5
DYNAMIC STABILITY, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

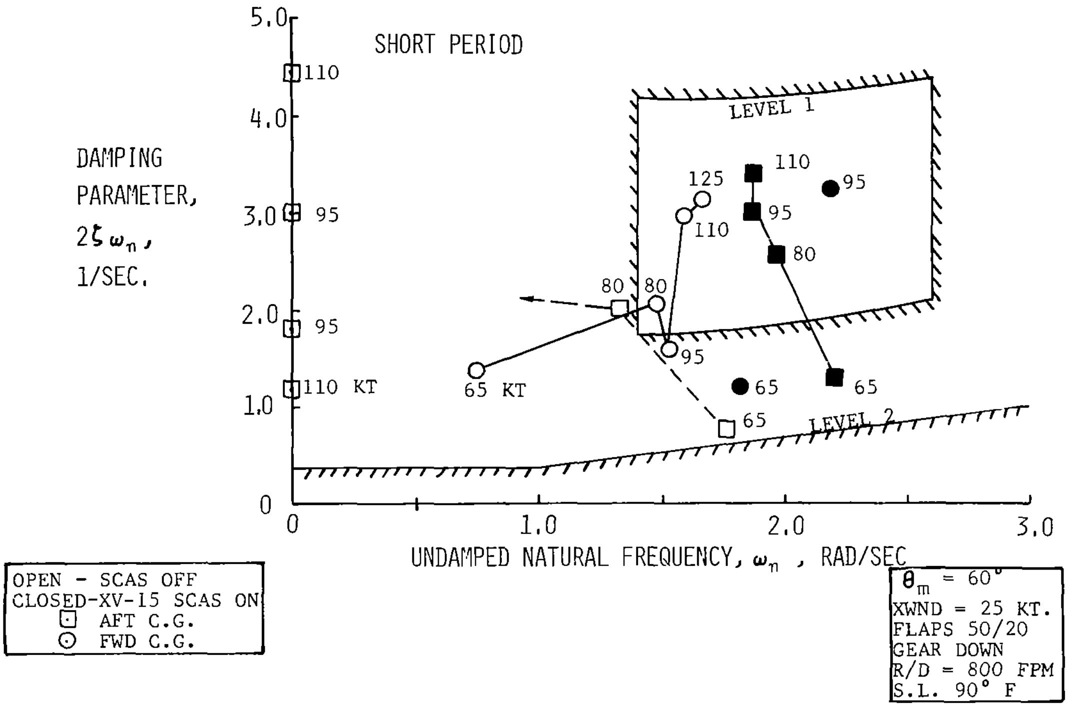
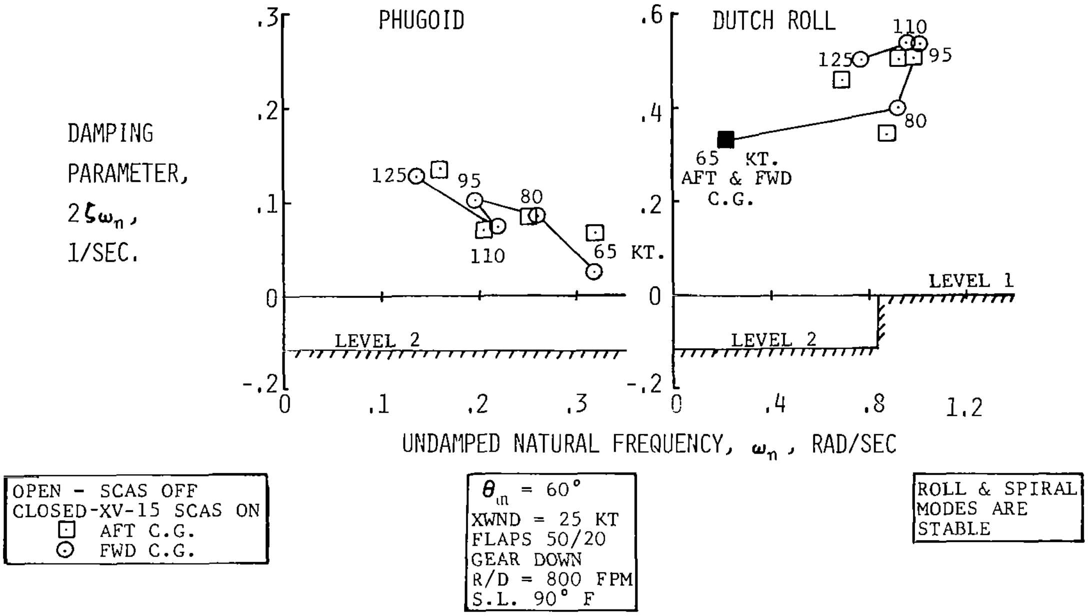


FIGURE 7.4-6
DYNAMIC STABILITY, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



aircraft (SCAS-off) meet the Design Criteria Level 1. With the exception of the 160-knot (297 kph) point at forward c.g., the Dutch Roll mode exceeds the requirements of the AGARD-R-577-70 Normal Flight limit; however, the 160-knot point does meet the AGARD Single Failure limit. Above 180 knots (334 kph), the Dutch Roll mode also meets MIL-F-8785B Level 1, Category B damping and frequency requirements.

In the landing approach configuration (Figures 7.4-5 and 7.4-6), the results for each of the three stability modes are similar to those for the takeoff climb configuration. The Level 2 conditions are satisfied for the Short Period mode at either c.g. location at the minimum control speed (65 knots, 120 kph). The basic aircraft possesses sufficient Phugoid damping and again, SCAS would be necessary to damp the Dutch Roll mode below 80 knots (148 kph).

7.5 CONTROL POWER AND ATTITUDE RESPONSE

Attitude control power (determined from the trimmed cockpit control positions, total available control moment sensitivities from the rotor and control surfaces, and the appropriate inertias) are analyzed for four different flight configurations as shown in Figures 7.5-1 through 7.5-4. The study is made in each of the three principal axes (single channels) for the appropriate mission profile conditions to determine the most critical condition which would satisfy the minimum Level 1 requirements of Design Criteria paragraph 1.1.1. Yaw control power is obtained from the rudder, differential cyclic pitch and differential collective pitch on the rotor. Roll control is obtained from asymmetrical outboard spoiler deflection and differential collective pitch on the rotor. Pitch control is obtained from the elevator and symmetrical cyclic pitch on the rotor. Thrust/lift control is provided by symmetrical collective pitch on the rotor. In landing configuration, symmetrical spoiler deflection is also used for lift control.

Available angular accelerations in each axis for the takeoff-climb configuration with a 25-knot (46 kph) crosswind are shown in Figure 7.5-1. Control power at both c.g. extremes in each axis is sufficient to meet the requirements for trimming in a climb (with crosswind) from minimum control speed to 125 knots (232 kph); and subsequently, for possessing enough control margin to accelerate beyond the minimum Level 1 criteria by applying the appropriate control independently from trim to the nearest stop. In addition, the available pitch control power was investigated during a takeoff ground roll at 80 knots with the gear still in contact with the runway. It was found that sufficient control power exists to rotate the aircraft about the main gear to the desired liftoff attitude and retain an aft stick margin.

FIGURE 7.5-1
SINGLE-CHANNEL CONTROL POWER, TAKE-OFF CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

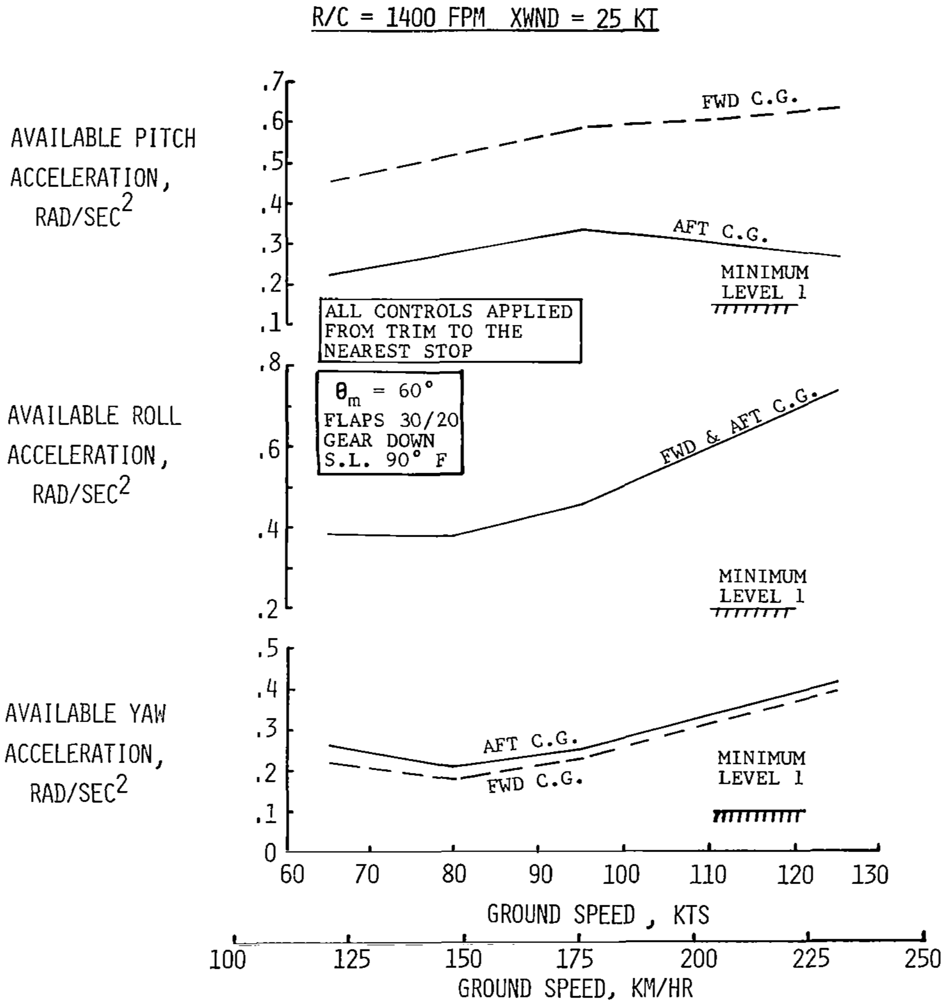


FIGURE 7.5-2
SINGLE-CHANNEL CONTROL POWER, CONVERSION CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

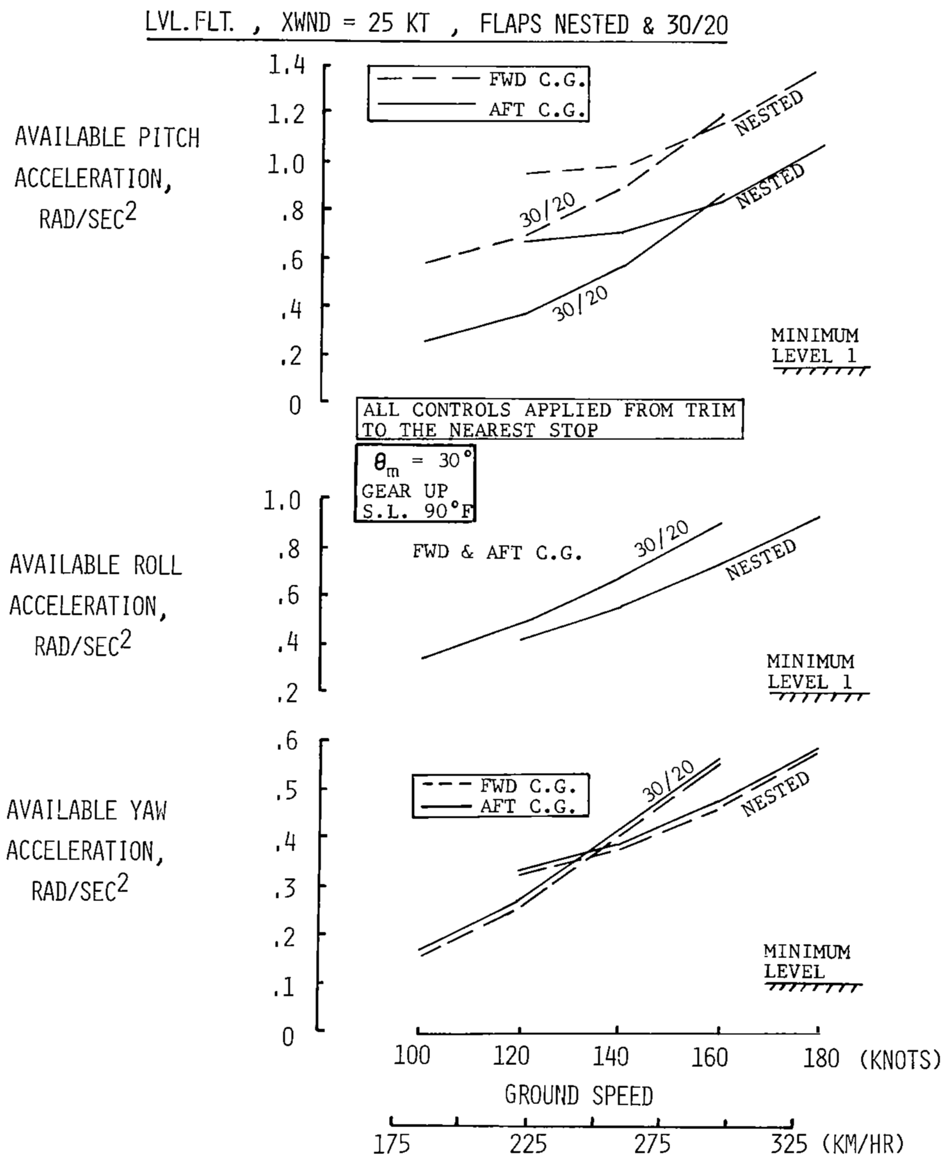


FIGURE 7.5-3
SINGLE-CHANNEL CONTROL POWER, AIRPLANE CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

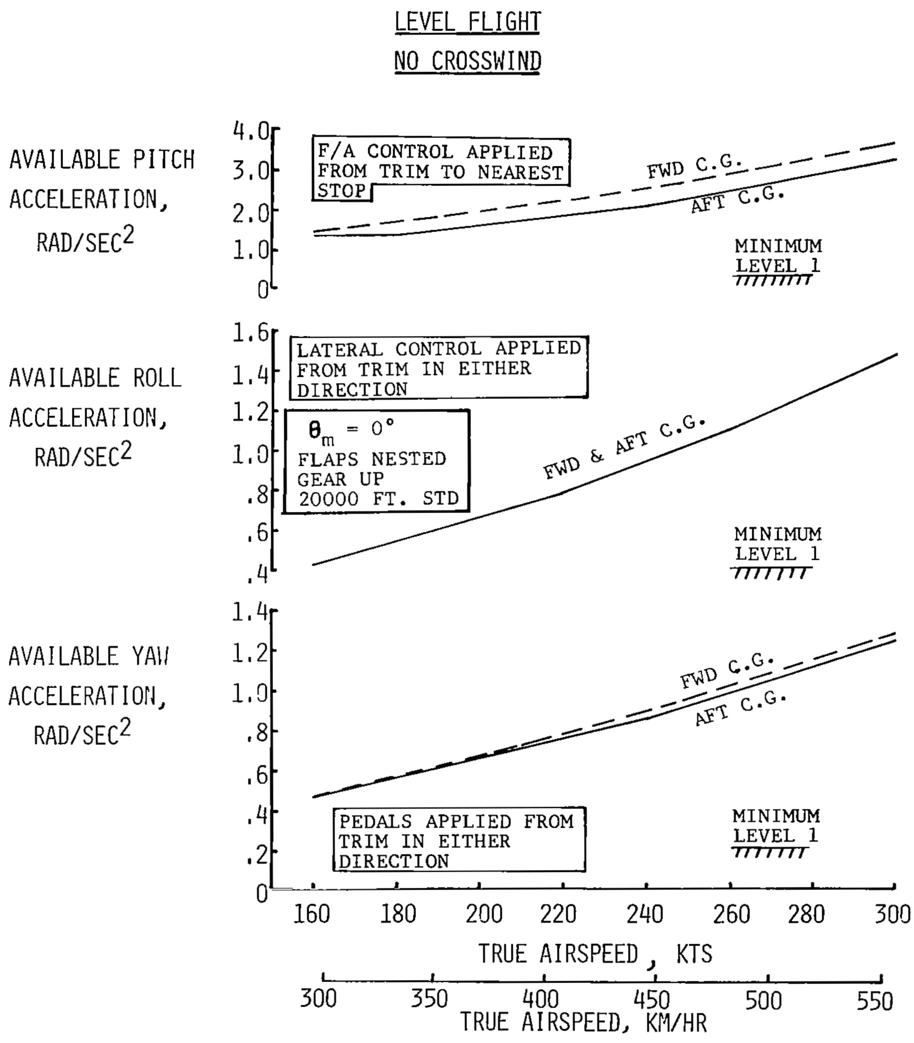
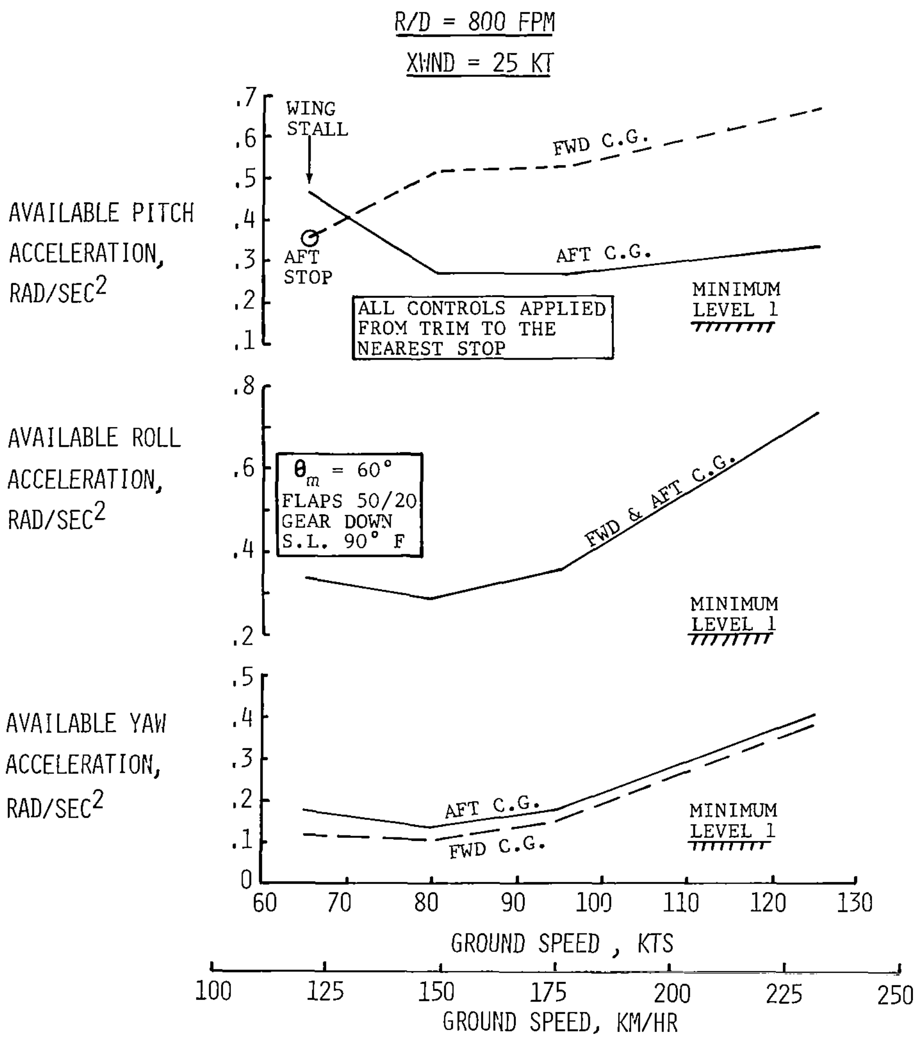


FIGURE 7.5-4
SINGLE-CHANNEL CONTROL POWER, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



Attitude control power for the conversion configuration ($\theta_M = 60^o$) in level flight with a crosswind is presented for both the flaps nested and 30/20 positions (Figure 7.5-2). Adequate acceleration capability from trim exists for all conditions shown.

Figure 7.5-3 shows the control power capabilities for the airplane cruise mode at 20000 feet (6096 m), standard day, level flight, no crosswind conditions. Since the trimmed conditions are with neutral pedals and lateral stick, these controls can be applied to either stop to obtain the designated accelerations. The most critical roll acceleration condition exists at 160 knots (297 kph) which is only 8% above stall speed. If necessary, additional roll control could be obtained from deflection of the inboard spoiler panel. However, since this point is below 1.2 x V stall and the minimum Level 1 criteria (.4 rad/sec^2) is met, the condition is considered adequate.

In the landing descent configuration (Figure 7.5-4), the most critical yaw acceleration condition exists at 80 knots (148 kph) and forward center-of-gravity. In this crosswind condition, the remaining yaw control power is exactly .1 rad/sec^2 which is the minimum Level 1 requirement (50% of .2 rad/sec^2). The pitch control power below 80 knots (148 kph) indicates a reversal in trend for both c.g. conditions due to the stick positions resulting from wing stall (discussed in Section 7.3). The nearest stop is the aft limit at forward c.g., thereby reducing the available margin in trim. The forward margin is significantly increased at the aft c.g. condition over that which would exist if the wing were not stalled.

In order to demonstrate the maneuver control power capability following the simultaneous application of all three primary controls, Table 7.5-1 shows an example at the most critical yaw condition discussed above. Along with the left pedal input, the stick is simultaneously stepped to the left/forward limit. Because of the moderate amount of rotor roll/yaw coupling, the yaw acceleration is more than twice the requirement (100%) and the resulting roll and pitch acceleration capabilities far exceed their requirements (30%).

Time histories of yaw, pitch and roll attitude response to independent control inputs at the most critical acceleration conditions discussed previously are shown in Figure 7.5-5. The yaw and roll critical conditions are discussed above while the most critical condition for pitch acceleration exists in the landing approach mode at minimum control speed and aft c.g. Although the available acceleration capability is acceptable at each one of the points, these represent the most critical conditions in each axis and, therefore, were selected to

TABLE 7.5-1
SIMULTANEOUS CONTROL POWER, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

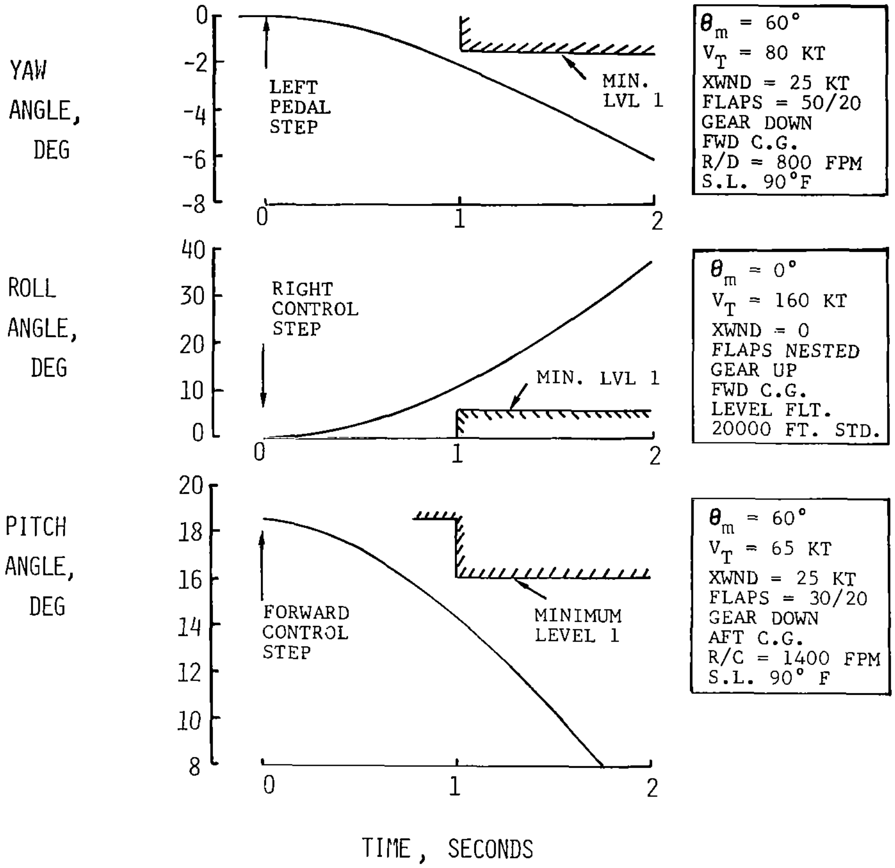
R/D = 800 FPM, XWND = 25 KT
GROUND SPEED = 80 KT

	YAW ACCELERATION	ROLL ACCELERATION	PITCH ACCELERATION
DESIGN CRITERIA (PARAGRAPH 1.1.1, CONDITION (b))	100%	30%	30%
REQUIREMENT (RAD/SEC^2)	±.20	±.12	±.09
CAPABILITY (RAD/SEC^2)	-.42	-1.19	-.57

FORWARD C.G.
$\theta_M = 60°$
FLAPS 50/20
GEAR DOWN
S.L. 90° F.

LEFT PEDAL AND
LEFT/FORWARD
STICK SIMULTANEOUSLY
APPLIED FROM TRIM
TO THE STOPS

FIGURE 7.5-5
ATTITUDE RESPONSE, SCAS OFF
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



indicate the angular response within one second following a step input. The attitudes exceed the minimum requirements in all three axes.

7.6 CRUISE FLIGHT MANEUVER STABILITY

The stick-fixed maneuver stability at mission cruise conditions and design gross weight is shown in Figure 7.6-1. The forward center-of-gravity point (FS 485.7) is located at 16.75% MAC while the aft center-of-gravity point (FS 504.2) is located at 33.14% MAC. This c.g. range represents a payload shift of ±5% of the cabin length. Both of these limits possess positive maneuver stability without the use of SCAS. The stick-fixed maneuver point, i.e., that c.g. location at which the elevator deflection per g level equals zero, is located at 57.74% MAC (FS 532), providing a maneuver margin in this flight regime for the aft c.g. of 24.6% MAC (27.8 inches, .71 m).

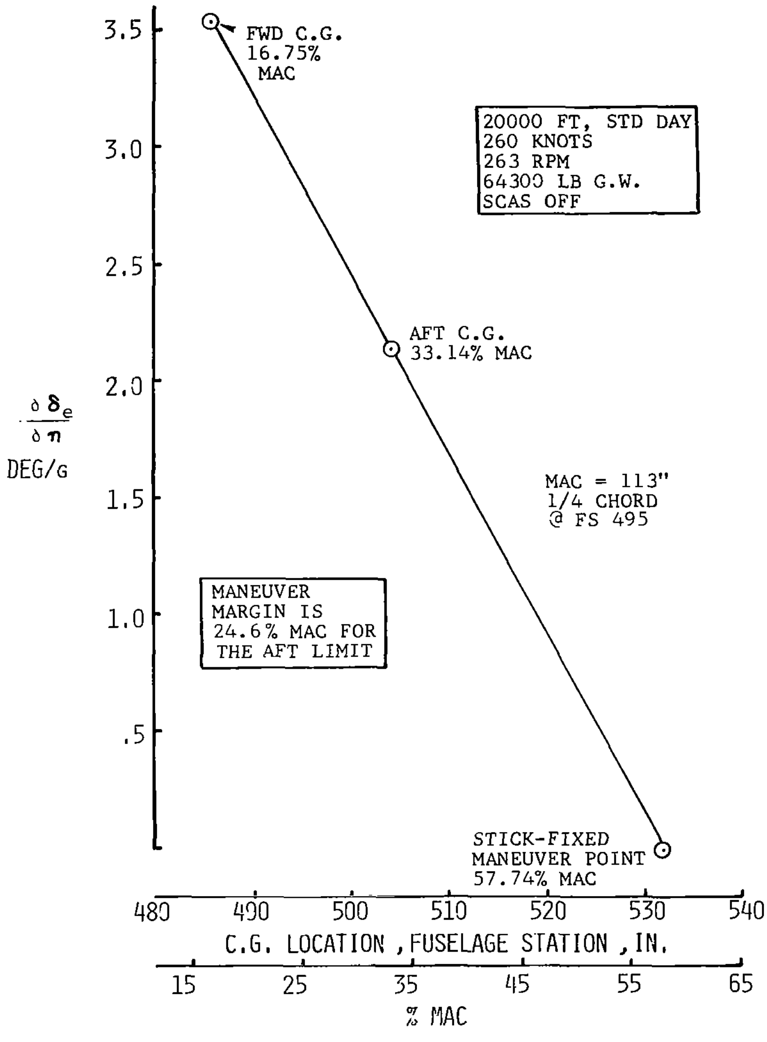
Using the current XV-15 force-feel constants (a stick-force gradient of 15.5 lbf/in. (27.14 n/cm) at 260 knots (482 kph)) provides values of 13.17 and 7.96 lbf (58.6 and 35.4 N) per g for the forward and aft c.g. limits, respectively. These results indicate that the center-of-gravity envelope could be extended beyond that studied. Any envelope expansion would also have to be within the limits of the static stability margins (Figures 7.3-1 and 7.3-2), available pitch control power (Figures 7.5-1 through 7.5-4), and the SCAS-on dynamic stability requirements.

7.7 EMPENNAGE SIZING

Empennage sizing and directional and longitudinal characteristics are discussed in the following sections.

7.7.1 DIRECTIONAL CHARACTERISTICS - During the analysis of this STOL configuration, an investigation was undertaken to determine a solution for increasing the yaw acceleration capability in a crosswind. This effort was recommended in the Appendix of Volume 1 of this study (Reference 2-2). The primary conclusion is that if the vertical tail volume coefficient is reduced from the initial baseline value of .2 down to a value of .13 (by reducing the fin area), then the yaw moment with sideslip of the total aircraft (N_β) will be reduced in a crosswind and thereby result in an increased pedal margin in trim. In addition, three different control rigging changes are made as follows: rudder chord is increased from 25% (XV-15) to 40% of the fin chord and full rudder travel is increased from ±20° (XV-15) to ±25° by a rigging change; differential collective pitch is rigged to the pedals as well as the lateral stick for the purpose of increased yaw control power; and approximately the same differential cyclic travel

FIGURE 7.6-1
MANEUVER STABILITY, AIRPLANE CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



with pedal is available during an STOL takeoff or landing (θ_M = 60°, V_T = 80 knots (148 kph)) as the XV-15 in hover with the mast vertical.

The combination of reduced vertical tail coefficient and the three control modifications produces the following results. The yaw acceleration requirement in a crosswind condition (Design Criteria paragraph 1.1.1) is satisfied throughout the STOL flight envelope. Also, the Dutch Roll damping and frequency requirements of both the Design Criteria (paragraph 1.1.4) and AGARD-R-577-70 are satisfied in the airplane mode cruise altitude/low speed condition (see paragraph 7.4). The only disadvantages of the above modifications are the increased control system complexity (more so in a mechanical system than a fly-by-wire system) and that the MIL-F-8785B Level 1 Dutch Roll damping requirement is not satisfied between wing stall speed (148 knots, 274 kph) and 180 knots (334 kph) in the airplane cruise mode. However, these are considered minor compared to the gains in the D313 yaw acceleration capabilities over those of the D312 described in Volume 1 of this report. It should be noted that these modifications are also applicable to the VTOL tilt rotor.

7.7.2 LONGITUDINAL CHARACTERISTICS - An increase in the horizontal tail volume coefficient from the initial baseline value of 1.15 (XV-15 value) to 1.639, by increasing the stabilizer area, is necessary in order to compensate for the much larger volume and destabilizing pitching moment with angle-of-attack value of the 100-passenger fuselage over that of the XV-15 fuselage. This C_H value of 1.639 used in the study produced positive static stability throughout the flight envelope. With regard to dynamic stability, a somewhat larger tail volume coefficient of 1.91 would be necessary to eliminate the split aperiodic Short Period roots of the basic aircraft (SCAS-off) in the airplane mode. However, as discussed in Section 7.4, a SCAS could be designed to accomplish the same task of meeting the Level 1 Short Period criteria.

8. AEROELASTIC STABILITY

An important design requirement for the tilt rotor is the provision of adequate aeroelastic stability margins of the rotor-wing combination for the speed-altitude envelope capability of the aircraft. The D313 STOL aircraft has identical rotors and spanwise locations (relative to the fuselage) as the D312 VTOL. The D312 rotor-wing aeroelastic stability was verified and reported in Reference 2-2; however, the design changes to the wing aspect-ratio and sweep angle required that the D313 be verified also.

Three sizing cases were considered in the synthesis of the D313 wing:

a. Helicopter Bending - This is the maximum transient static thrust which can be produced by the rotor. It is applied with the mast vertical to determine the wing root moment. For the D313 it is 1.613 times the design static thrust.

b. Airplane Bending - This is the wing root moment developed during the application of the design normal load factor in airplane cruise mode. For the D313 it is 2.5g, with the lift vector placed (conservatively) at the mid-point of each semi-span.

c. Torsion - This is the wing torsional stiffness required to produce a design torsional rotor-to-wing frequency ratio for a given pylon weight and offset from the wing torsional axis. For the D313 the wing design torsional frequency requirement was ≥ 0.8/cycles per rotor revolution in airplane mode.

The D313 critical case was case b. "airplane bending." Thus the wing section required for the conventional airplane bending exceeded the requirements of cases a. and c. This section was then analyzed using BHC computer program DYN4 and a description follows.

8.1 METHOD ANALYSIS

The parameters defining kinematics and structural quantities were generally obtained from the Tilt Rotor Aircraft Design Synthesis program (OMSW03). The parameters of wingtip beamwise spring rate, chordwise spring rate, wing effective mass, wind chord effective hinge location, pylon pitch and yaw spring rates were scaled from the XV-15. Pylon center-of-gravity and pitch inertia was recalculated for the D313 conversion axis location, forward of the front wing spar. The aircraft rigid

body stability derivatives were calculated using Munk's method. Studies of aeroelastic stability were made by treating symmetric modes about the fuselage longitudinal centerline separately from those antisymmetrical about the centerline. For the symmetric or antisymmetric modes, the DYN4 math model consists of the following degrees-of-freedom.

a. Two rigid-body flapping modes, one involving backward precession in the rotating system; the other, forward precession. These are both symmetric and antisymmetric modes.

b. Three rigid-body airframe modes: plunging, pitching and longitudinal translation in the symmetric case; and roll, yaw, and lateral translation in the antisymmetric case.

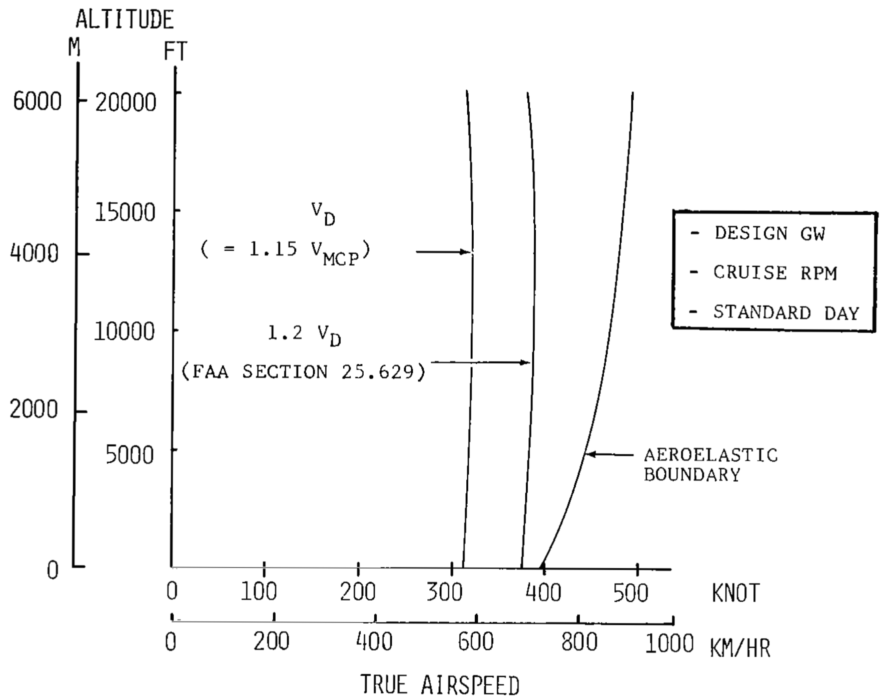
c. Five wing-pylon elastic degrees of freedom: wing beamwise bending, chordwise bending, and torsion; and pylon pitch and yaw with respect to the wing. These are for both symmetric and antisymmetric modes.

These ten degrees-of-freedom for each set of modes, which are completely coupled in the analysis, were considered to be adequate to represent the coupled natural modes of the D313.

8.2 RESULTS

The criterion for aeroelastic stability for the commercial transport is taken from the FAA Airworthiness Standards: Transport Category Airplanes, Part 25, Section 25.629 (Reference 8-1). FAR Part 25 requires that the aircraft be designed to be free from flutter and divergence for all combinations of altitude and speed encompassed by the dive speed (V_D) versus altitude envelope, enlarged by an increase of 20% in equivalent airspeed. Based on this criterion, and defining V_D as 1.15 times the speed at maximum continuous power, V_{MCP}, the D313 has sufficient margins for aeroelastic stability, Figure 8.2-1.

FIGURE 8.2-1
AEROELASTIC MARGINS
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



9. SAFETY ASPECTS

This section covers the safety aspects of one-engine-out performance, low-speed gust response, and critical component redundancy.

9.1 ONE-ENGINE-INOPERATIVE PERFORMANCE IN THE STOL MODE

Since the rotors are mechanically interconnected and any engine can drive either rotor, there is no critical engine or asymmetric thrust condition, Engine-out performance in the takeoff configuration is shown in Figure 5.4-1. Rate of climb with three engines at IRP (30-minute rating) at sea level 90°F (32.2°C) is indicated by the 75% power contour and is 1060 ft/min (323 m/min) at 80 knots (148 kph). At the 2½-minute rating the rate of climb is indicated by the 90% power contour (transmission limit) and is 1520 ft/min (463 m/min) at 80 knots (148 kph). These capabilities exceed the FAA Part XX engine-out requirement of 250 ft/min (76.2 m/min).

9.2 ONE-ENGINE-INOPERATIVE PERFORMANCE IN THE AIRPLANE MODE

One-engine-out performance in airplane mode is shown in Figure 9.2-1. Cruise speed on three engines, at maximum continuous power, exceeds best-climb speeds (1.2 V stall) throughout the flight envelope. Speed capability is 235 knots (435 kph) at 7000 feet (2134 m).

9.3 LOW-SPEED GUST RESPONSE

Aircraft response to four discrete sharp-edged gusts during a typical STOL final approach to landing is presented in Figure 9.3-1 (longitudinal gust) and Figure 9.3-2 (lateral gust). These horizontal gusts are of 15 fps (4.6 m/sec) amplitude for a duration of 5 seconds, originating longitudinally from the forward and aft directions, and laterally from the left and right, in the earth-based coordinate system. The aircraft is initially trimmed (at the 2-second point) in a 25-knot (46 kph) steady-state crosswind from the right with an 800 fpm (244 m/min) descent rate and an 80-knot (148 kph) forward (ground reference) speed with the flaps 50°/20° and the gear down. This descent condition is the same as that shown in Figures 7.3-2, 7.4-5, -6, and 7.5-4.

No corrective action by the pilot or SCAS is present during these response analyses. All of the longitudinal and lateral/directional stability modes at this aft c.g. condition (see Figures 7.4-5 and 7.4-6) are stable without the use of SCAS. In each case it can be seen that the basic aircraft has sufficient attitude and velocity damping to continue sustained

FIGURE 9.2-1
CRUISE PERFORMANCE, ONE ENGINE INOPERATIVE
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

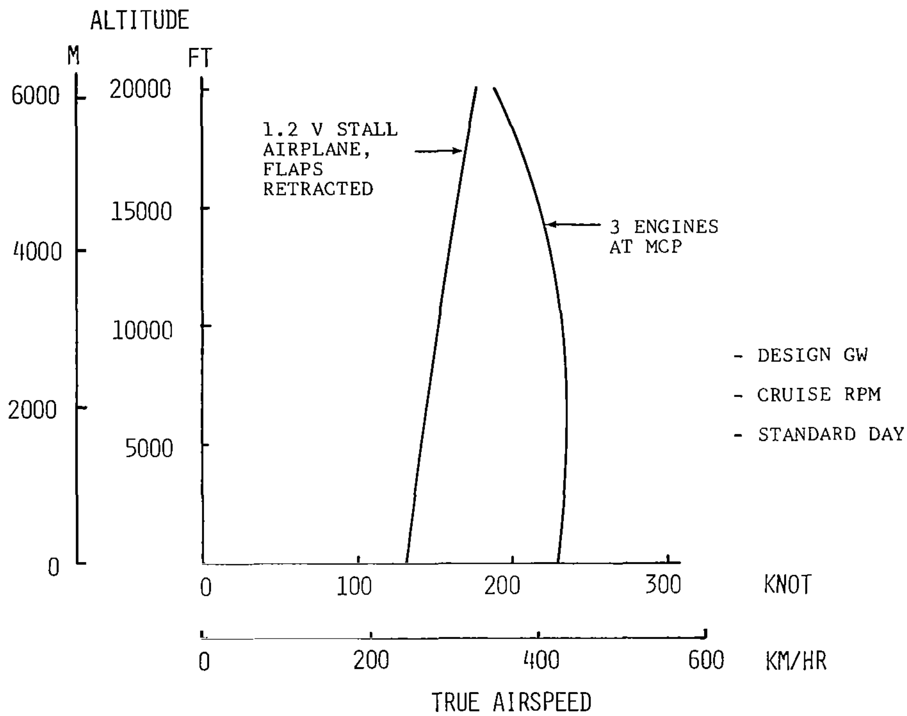


FIGURE 9.3-1
RESPONSE TO LONGITUDINAL GUST, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF

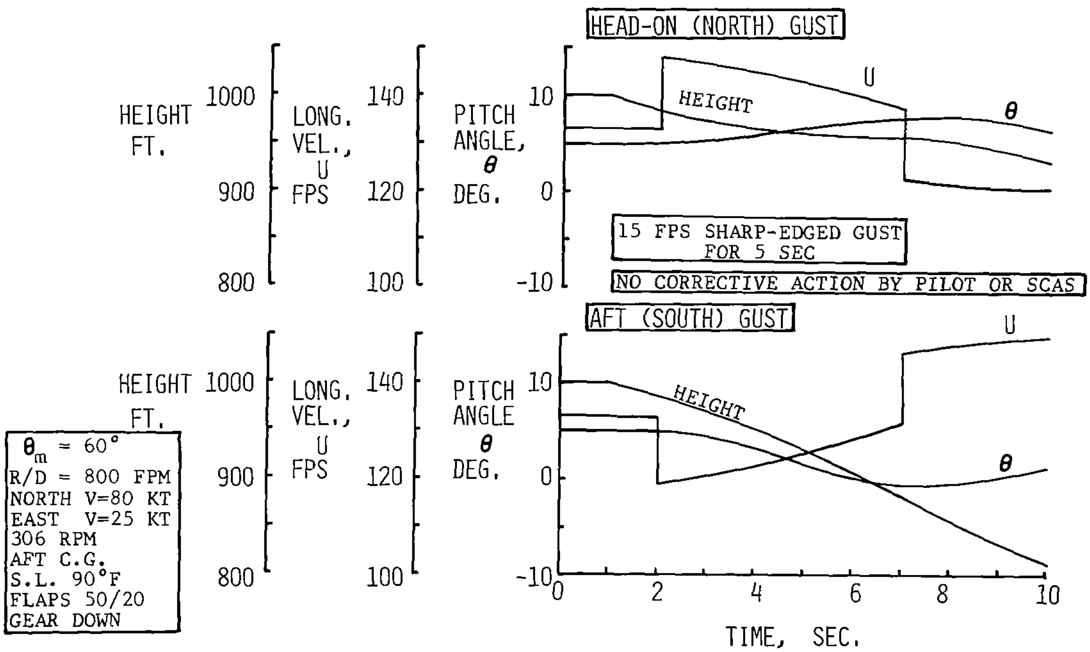
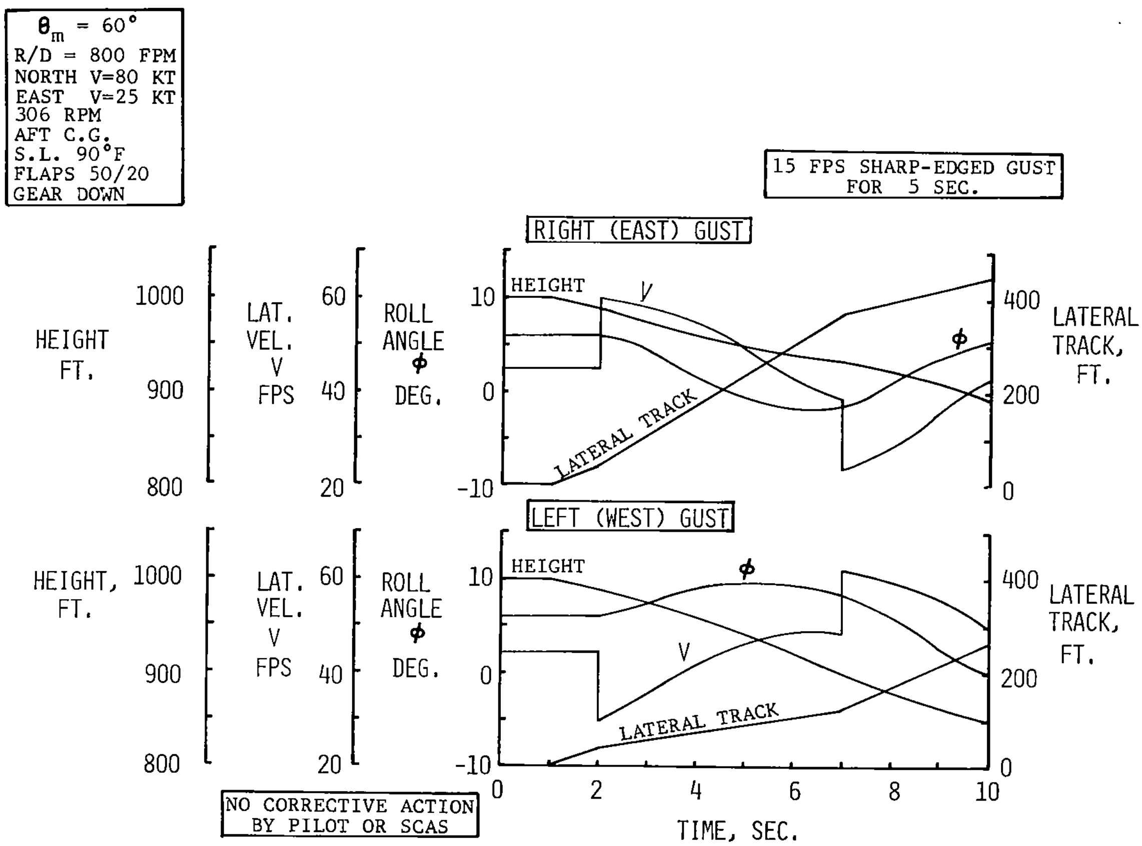


FIGURE 9.3-2
RESPONSE TO LATERAL GUST, LANDING CONFIGURATION
100-PASSENGER STOL AIRCRAFT
DGW = 64300 LBF



flight without SCAS or pilot corrective action during the gust duration. Following the removal of the gust, some corrective action, by either the pilot or the SCAS, might be necessary to eliminate excessive pitch or roll attitudes. As with the VTOL configuration defined in Volume 1 of this report, additional investigations with pilot-in-the-loop simulation are recommended.

9.4 GENERAL SAFETY CHARACTERISTICS

The two low disc loading rotors at mast angles of 70°-90° provide autorotation capability for a reduced descent rate emergency landing in case of fuel exhaustion or total loss of power. Adequate collective pitch range and rotor solidity (total blade planform) permit rotor speed control during descent and provide flare thrust to reduce rate-of-sink. The landing gear is designed to withstand a vertical sink rate of 10 fps at the design gross weight.

The rotors are driven by wingtip-mounted turbine engines. An interconnecting shaft system between the rotors (cross-shafting) allows any engine to power both rotors in the event of an engine or engine gearing failure. Driving each of the rotors independently is also possible in the case of a cross-shaft failure. Rotor desynchronization due to a cross-shaft failure will not cause rotor intermeshing problems (as on some tandem helicopters) because the rotors do not overlap.

Overrunning clutches in the engine reduction-gearing automatically disconnect a failed engine from the drive system, thus allowing the effective use of available power. Redundant transmission housing mounting-lugs prevent a catastrophic single bolt or lug failure. The drive system strength requirements allow for uneven power distribution (such as a double-engine failure on one side) and maneuver or gust transient loads and torques. For normal operation, torque limitations will be placarded and are a pilot-control function.

The Bell stiff-in-plane tilt rotor design philosophy, as used for the XV-15, is considered to be a major design parameter to ensure flight safety. With an inherently stable dynamic system, a failure of the stability and control augmentation or a gust alleviation feedback-system will not lead to a catastrophic instability.

The conversion (nacelle tilt) mechanism is provided with dual hydraulic actuation and redundant control subsystems to enable full range operation after any single failure. In the event of two hydraulic failures, the nacelles can be converted by the use of a drive system powered by the utility hydraulic system. A nacelle synchronization feature is also provided.

Three separate hydraulic systems would be typically installed in a four-engine transport; two primary flight control systems and a utility system. The primary systems would be powered by a hydraulic pump driven from each main rotor transmission. The utility system would be powered by a hydraulic pump driven from the interconnect shaft, adjacent to the fuselage, so that hydraulic power is available as long as the rotors are rotating. In addition, the auxiliary power unit (APU) drives the utility pumps which would power the utility system for ground check-out.

Critical components of the separate systems will be physically isolated, where possible, to prevent concurrent failure due to local damage. The flight controls will be fly-by-wire and include force-feel and stability and control augmentation functions. Controls that are not safety-of-flight items may be powered by single actuators. Built-in test equipment (BITE) will be provided. Fire resistant hydraulic fluid will be used to reduce the fire potential of the hydraulic system.

The electrical system follows the same design approach as for the hydraulics; three completely independent systems, of which one generator is driven by each rotor transmission and the remaining generator by the interconnect shaft. In addition, the APU and the batteries provide electrical power on the ground and as desired by the pilot in flight. Adequate electrical power for the critical flight-required equipment will be available after the loss of any two of the electrical systems.

An engine fire detection and pilot actuated fire extinguishing system will be incorporated. Engine inlet icing detection and anti-icing are also provided. Fuel is stored in the wings, outboard of the fuselage, in integral spray-in cells. Break-away fittings are utilized to eliminate fuel spillage from fuel lines separated in a crash. The remote location of the engines from the fuselage reduce the hazard, to the passengers and crew, of engine fire and the resulting smoke and heat.

Nose gear swiveling and differential braking are provided for ground operation. For the 100-passenger STOL aircraft, the lowest part of the rotor disc in the normal takeoff configuration (60° mast) will have over 10 feet of ground clearance. The crew members will have an unobstructed view of the out-board rotor tip path to reduce the hazard of rotor tip collision with ground objects during taxi or ground maneuvering.

Flight operation at takeoff and landing will display safety characteristics approaching helicopters because of the high

thrust/weight ratio (0.75) and the inertia of the rotor system. Control powers and sensitivities are greater than the minimum levels recommended in AGARD Report No. 577.

Transition to cruise flight is performed within the boundaries established by wing stall, the torque limit, or rotor/hub endurance limits. The allowable corridor is broad (generally greater than 90 knots).

The general flight characteristics in cruise are those of a turboprop airplane. Conventional aircraft control surfaces are employed.

A pilot caution and warning system will provide visual and/or audible indications of detectable system malfunctions, such as hydraulic system pressure loss, rotor control discrepancies, engine fire, etc.

10. CONCLUSIONS

A conceptual design study of a 1985 commercial STOL (2000-foot, 610-m field length) tilt rotor transport, based on a NASA STOL mission, has been completed. The conclusions are as follows:

1. The STOL variant (D313) of the 45-passenger VTOL tilt rotor can fly the NASA 200 n. mi. mission at significantly reduced DOC and increased fuel economy. The payload increased 122 percent to 100 passengers; the DOC reduced 43 percent to 2.67 ¢/assm (1.66 ¢/askm) and the fuel economy increased 137 percent to 81.1 seat-statute miles per gallon (29.6 seat-kilometers per liter).

2. Compared to present (1975) CTOL aircraft such as the B-737, the 100-passenger D313 uses 38 percent of the B-737 fuel for the 200 n. mi. (370 km) mission.

3. Compared to 100-passenger CTOL fan-jet short-haul aircraft of the same time frame (operational in 1985), which are estimated to achieve 40 ssmpg (fan bypass ratios 8-10), the 100-passenger 1985 STOL tilt rotor is estimated to have a lower DOC for fuel costs above 10¢/lb. The tilt rotor mission time for 200 n. mi. (370 km) is calculated to be 54 minutes compared to 48 minutes for the CTOL.

4. The STOL tilt rotor inherits the control capabilities of the VTOL tilt rotor. In severe crosswinds of up to 25 knots during an 80-knot approach, the STOL tilt rotor can achieve a unique zero-bank, zero-crab approach, to reduce pilot workload. This feature can be investigated during the flight testing of the NASA-Army XV-15 tilt rotor aircraft, provided a normal range of lateral cyclic is available.

5. A high degree of commonality exists between the 45-passenger VTOL tilt rotor (Bell D312) and the 100-passenger STOL tilt rotor (Bell D313). The rotors, main transmissions, and engines are identical. This would enable a commercial airline to operate VTOL and STOL aircraft over a wide combination of missions and at ranges from 100 to 1000 n. mi. (185-1850 km).

6. Achieving the predicted characteristics of the STOL D313 tilt rotor is dependent on the applicable technology programs taking place in the 1976-1979 time period. These include tilt rotor flight simulation, flight research with the XV-15 and advanced technology components.

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APPENDIX

The fuel cost specified by NASA in this study was 2¢/lb, which was typical for 1972-73 U.S. domestic airline operations. The rapid increase since then, to 4-5¢/lb in mid-1975 and an anticipated 10¢/lb in 1980-85, compelled a brief study of this cost impact.

Aircraft solutions were synthesized at fuel costs of 10¢/lb and it was found that minimum DOC solutions occurred at higher cruise speeds than that of the D313 (which was optimized for maximum fuel economy). Accordingly, installed power was increased until the cruise speed (at 90% MCP) for minimum mission-DOC was defined. Also, a twin-engine design was found to weigh less than a four-engine design; and with the higher installed power, a twin-engine design was found to have adequate engine-out performance.

Results are summarized in Table A-1. For the 45-passenger class, a twin-engine aircraft (Point Design #15) with a cruise speed of 268 knots (496 kph) achieved a lower DOC than the four-engine design (Point Design #11) with a cruise speed of 234 knots (433 kph). Point Design #15 is considered to be quite competitive with current (and future) versions of 45-passenger class turboprop STOL aircraft.

For the 100-passenger class, a twin-engine aircraft (Point Design #19) with a cruise speed of 300 knots (556 kph) achieved a lower DOC than the most fuel-conservative four-engine design (D313, Point Design #2) with a cruise speed of 248 knots (459 kph). The DOC of Point Design #19 is estimated to be comparable (within 3%) to 1985 turbofan CTOL short haul transports (fan bypass ratio 8-10, fuel economy 40 ssmpg), Figure A-1, and to be more economical at fuel costs above 10¢/lb. As fuel costs increase beyond 10¢/lb, the cruise speed for minimum DOC will fall below 300 knots and approach that of the D313.

The mission fuel economy index versus design cruise speed for the 100-passenger STOL tilt rotor aircraft class is shown in Figure A-2. The curve defines the fuel economy index for a design solution at each cruise speed, with mission takeoff at Design Gross Weight, and cruise at 99-percent of maximum specific range. The D313 cruising at 248 knots (459 kph) achieves 81.1 ssmpg. At higher speeds, fuel economy decreases and at 300 knots (556 kph) the design solution for the mission achieves 72.0 ssmpg. This curve thus shows estimated generic characteristics and may be used for comparison with other aircraft classes.

FIGURE A-1
DIRECT OPERATING COST VERSUS FUEL COST

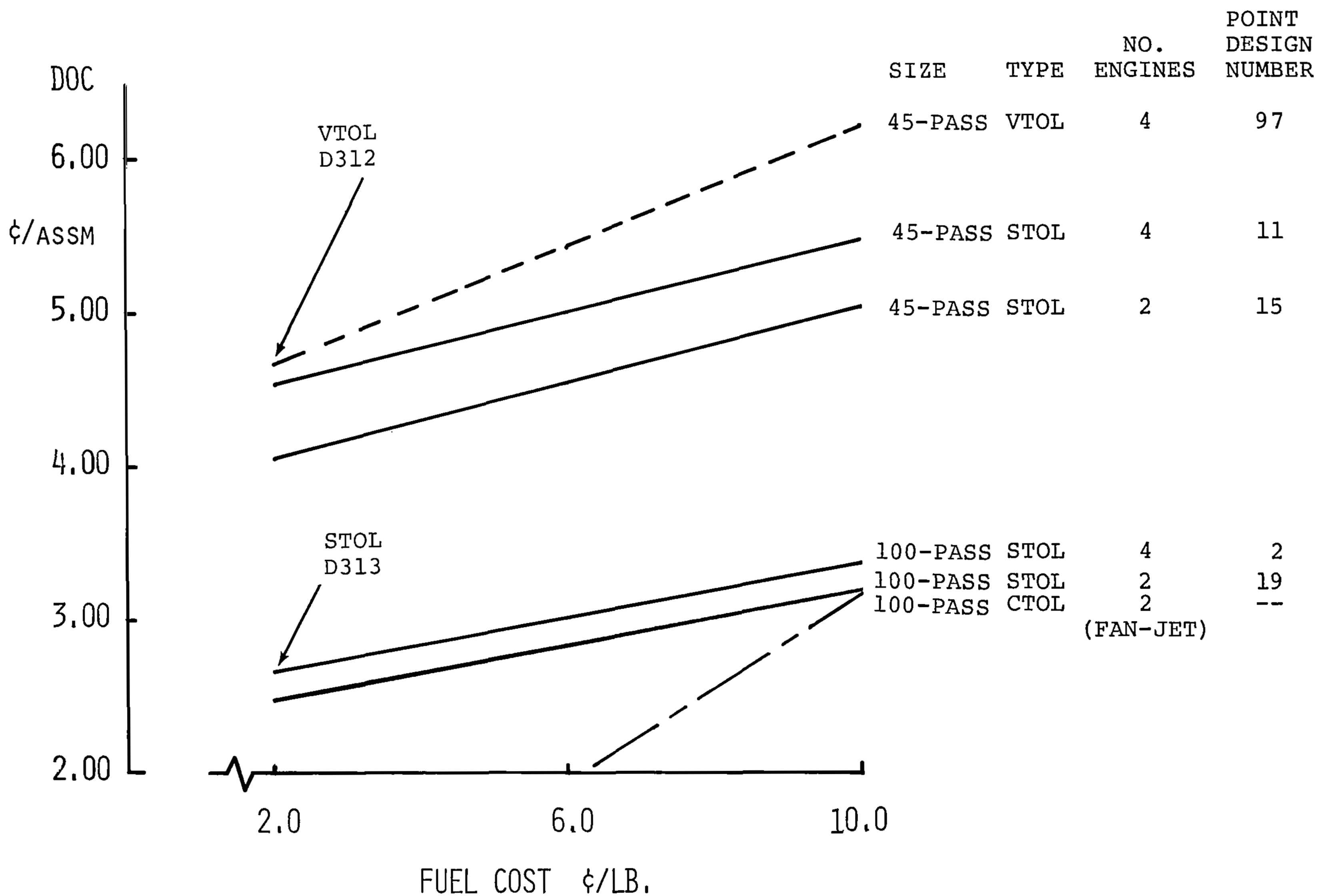
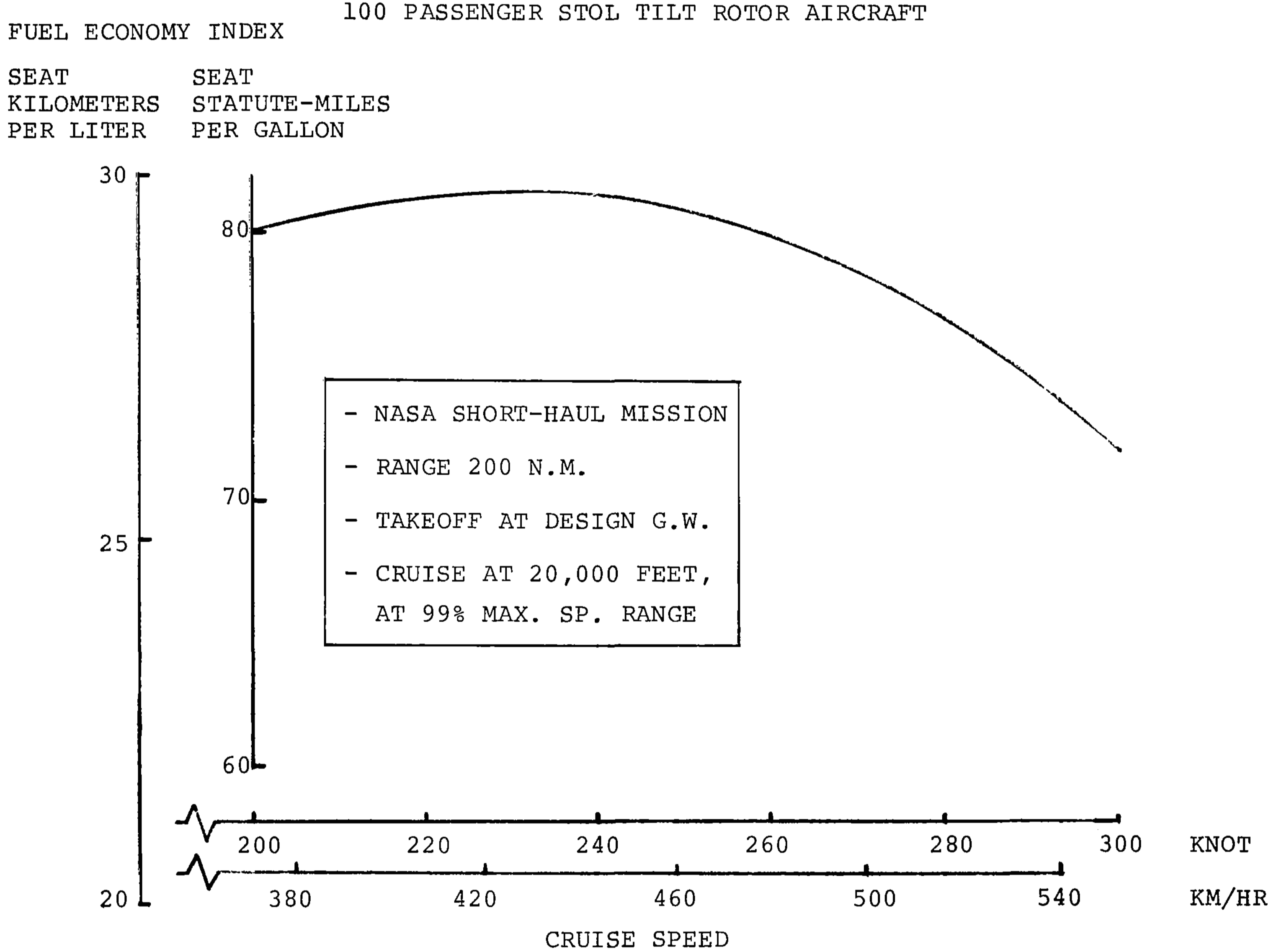


TABLE A-1

TILT ROTOR STOL AIRCRAFT FOR NASA 200 N.MI. SHORT-HAUL MISSION
FUEL COST = 10¢/LB (66.5¢/U.S. GALLON)

Point Design No.	DGW lb	Pass. No.	Eng. No.	Total Power Installed, hp at IRP,SLS	Rotor Dia. ft	Cruise Speed, knots	Fuel Economy ssmpg	DOC ¢/assm	Design Criteria
11	36975	45	4	5244	33.1	234	61.0	5.48	Minimum fuel
15	36360	45	2	6010	32.8	268	57.7	5.05	Minimum DOC
2	64300	100	4	9072	43.6	248	81.1	3.38	Minimum fuel
19	64850	100	2	12010	43.6	300	70.4	3.24	Minimum DOC

FIGURE A-2
MISSION FUEL ECONOMY VERSUS CRUISE SPEED,
100 PASSENGER STOL TILT ROTOR AIRCRAFT



NASA-Langley, 1976